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USATECOM PROJECT NO. 4-6-0500-01
USAAVSCOM PROJECT NO. 66-06
USAASTA PROJECT NO. 66-06

ENGINEERING FLIGHT TEST

AH-IG HELICOPTER HUEYCOBRA

PHASE B

PART 6

FINAL REPORT

RODGER L. FINNESTEAD
PROJECT ENGINEER

WILLIAM J. CONNOR
CWO, AV
US ARMY
PROJECT OFFICER/PILOT

NOVEMBER 1969

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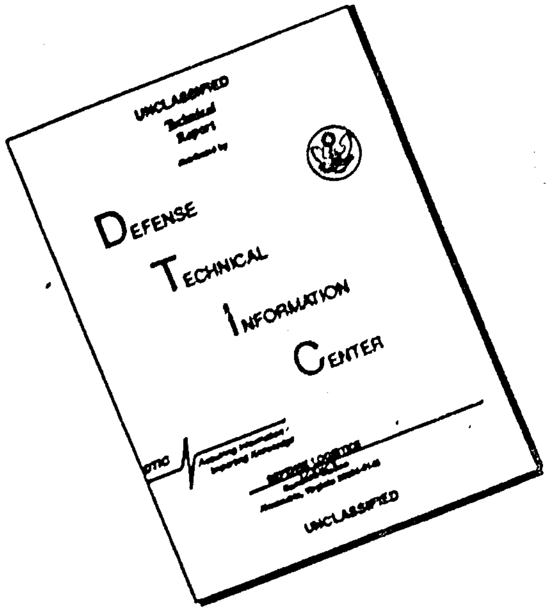

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US ARMY AVIATION SYSTEMS TEST ACTIVITY
EDWARDS AIR FORCE BASE, CALIFORNIA 93523

ABSTRACT

The AH-1G helicopter Phase B, Part 6 test program was conducted at Shafter, California, and Edwards Air Force Base, California, from 12 March through 3 May 1968 by the US Army Aviation Systems Test Activity, Edwards Air Force Base, California. The program was conducted to determine level flight performance, autorotational performance, engine characteristics, armed helicopter mission capability and to evaluate the in-ground-effect (IGE) handling qualities with the canopy doors removed. The helicopter is directionally unstable when hovering IGE with either the doors on or off in winds of 9 to 13 knots for azimuth range from 160 to 260 degrees (clockwise from nose of aircraft). This instability is a major deficiency and detracts from the mission capability of the aircraft. Undue pilot attention is required to avoid overtorquing the main transmission during maneuvers requiring abrupt left-lateral cyclic inputs in forward flight. This overtorque condition will only occur below the critical altitude of the engine. Additional deficiencies and shortcomings have been published in previous reports. Sufficient performance data were not obtained to determine the guarantee compliance.

TABLE OF CONTENTS

INTRODUCTION

BACKGROUND

1. The US Army Aviation Systems Test Activity (USAASTA) was directed by the US Army Test and Evaluation Command (USATECOM) to perform an engineering flight evaluation of the AH-1G helicopter (ref 5, app I). This testing was planned to be accomplished using several test aircraft during different time periods. The results of the Phase B performance tests using aircraft S/N 66-15247 are presented in this report. Handling qualities, vibration characteristics, wing stores jettison capabilities and armament subsystem evaluation test results are presented in parts 1 through 5 of the AH-1G Phase B report.

TEST OBJECTIVES

2. The objectives of this test were as follows:

a. To provide quantitative flight test data to serve as a basis for an estimate of the degree to which the helicopter is suitable for its intended mission.

b. To define the helicopter deficiencies to allow early correction and to provide a basis for evaluation of changes incorporated to correct deficiencies.

c. To provide limited performance flight test data for incorporation into the operator's manual.

d. To evaluate directional control margin in ground effect (IGE) with the canopy doors removed.

DESCRIPTION

3. The AH-1G helicopter manufactured by Bell Helicopter Company was designed specifically to meet the US Army requirement for an interim armed helicopter. The helicopter provides for a crew of two, seated tandem. The main rotor system is a two-bladed, door-hinge type with the customary stabilizer bar removed and a conventional antitorque tail rotor located at the top of the vertical stabilizer. The AH-1G is equipped with a three axes stability and control augmentation system to improve helicopter handling qualities. The power plant is a Lycoming T53-L-13

turboshaft engine rated at 1400 shaft horsepower (shp) at sea level (SL) under standard day uninstalled conditions. The engine is derated to 1100 shp because of the maximum torque limit of the helicopter's main transmission. The engine is equipped with a particle separator to prevent small foreign objects from entering the engine. The distinctive features of the AH-1G are the narrow fuselage (36 in.), the stub midwing with four external store stations and the integral chin turret. The flight control system is a positive, irreversible, mechanical type with conventional helicopter controls in the pilot's (aft) cockpit. The copilot/gunner's controls in the forward cockpit consist of conventional antitorque pedals, sidearm collective and cyclic controls. An electrical force trim system is connected to the cyclic and directional controls to induce artificial feel and to provide positive control centering. The elevator is synchronized with the cyclic stick. The armament configuration is changed by varying the wing stores and chin turret configuration. The pilot can fire all weapons in the stowed position. The gunner/copilot operates the flexible turret and can also fire the wing stores in an emergency using a pilot override switch. The wing stores can be jettisoned by either the pilot or gunner in case of an emergency. The design gross weight (grwt) for the AH-1G is 6600 pounds. Basic aircraft data and operating limits are presented in appendix IV.

SCOPE OF TEST

4. Thirty-five flights totaling 47.6 hours were conducted during the AH-1G Phase B performance and handling qualities testing. Testing was conducted at Edwards Air Force Base (2300-foot elevation), and Shafter (520-foot elevation), California, from 12 March through 3 May 1968. These tests consisted primarily of level flight performance, autorotational performance and engine characteristic and directional control handling quality evaluation IGE. The configurations tested are listed in table 1.

Table 1. Configurations.

Configuration	Armament Subsystems
Clean	TAT-102A turret, wing store stations - clean
Basic	TAT-102A turret, one XM157 outboard each wing
Inboard alternate	TAT-102A turret, one XM159 inboard each wing
Outboard alternate	TAT-102A turret, one XM159 outboard each wing
Light scout	TAT-102A turret, one XM18 inboard each wing, one XM157 outboard each wing
Heavy scout	TAT-102A turret, one XM18 inboard each wing, one XM159 outboard each wing
Heavy hog	TAT-102A turret, two XM159 each wing

5. The test program was conducted within the limitations established by the USAAVSCOM AH-1G Safety-of-Flight Release issued by AMSAV-R-F on 1 April 1967.

6. The empty gross weight of the test aircraft in a clean configuration with test instrumentation installed was 5790 pounds with a cg location at 205.97 inches. The test aircraft empty weight without instrumentation installed is not available since some test instrumentation was installed by the contractor prior to aircraft delivery to USAASTA. However, aircraft S/N 66-15327 had a dry weight of 5595 pounds and longitudinal cg of 204.18 inches. Both aircraft were equipped with a TAT-102 chin turret.

METHODS OF TEST

7. The methods and data reduction procedures used in these tests are proven engineering flight test techniques and are described briefly in appendix V.

8. All flights were conducted and supported by USAASTA personnel. Tests were conducted in nonturbulent atmospheric conditions.

CHRONOLOGY

9. The chronology of this test report is as follows:

Flight test commenced	12 March 1968
Flight test completed	3 May 1968
Preliminary data submitted	20 May 1968
Draft report submitted	17 May 1969

RESULTS & DISCUSSION

GENERAL

10. This report presents the results of engineering flight test Phase B performance and handling qualities of the AH-1G helicopter. Performance tests were conducted to determine the level flight performance, autorotational performance and engine characteristics of the AH-1G helicopter. Directional control tests were conducted IGE to determine if there was any change in handling qualities with the canopy doors removed. Sufficient performance data were not obtained to determine the guarantee compliance stated in reference 3, appendix I.

AIRCRAFT CONTROL SYSTEM COMPLIANCE CHECK

11. Prior to testing, the rigging of the aircraft and engine control systems was checked for compliance with the appropriate US Army manuals. As new procedures were made available to USAASTA, the aircraft and engine rigging changes were accomplished with coordination through the contractor's technical representatives.

LEVEL FLIGHT PERFORMANCE

12. Level flight performance tests were conducted at the test conditions specified. All these tests were performed with the frangible fairings removed ("unfaired" condition). End plates were placed over the front of each rocket pod to simulate a loaded pod (aerodynamically) when inert rockets were not used to achieve the desired aircraft weight.

13. Base-line level flight performance was defined for the heavy hog configuration. The level flight speed-power polars for the heavy hog configuration are presented in figures 4 through 8, appendix II and summarized in nondimensional form in figures 2 and 3. One level flight was conducted at a specified thrust coefficient (C_T) of 49.00×10^{-4} at a forward cg for the other armament configurations presented in table 1 to determine the effect of different wing store combinations on power required. The level flight performance for the different wing armament configurations is presented in figures 9 through 14.

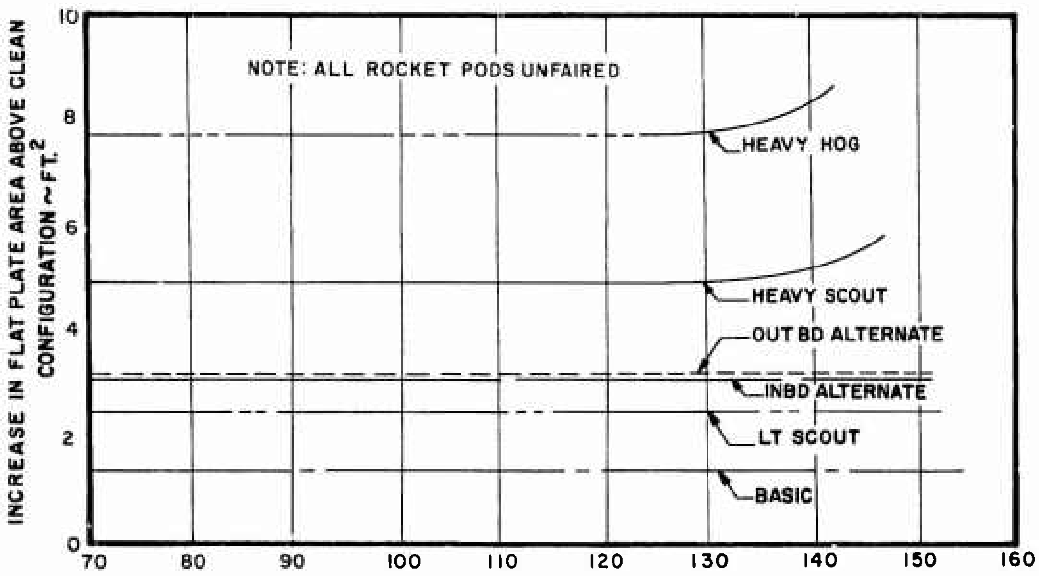
14. All subsequent configurations tested revealed an increase in equivalent flat plate area when compared to the clean configuration.

The increase in equivalent flat plate area for different configurations is presented in figure A for the thrust coefficient of 49.00 x 10^{-4}. The increase in equivalent area was greatest for the heavy hog configuration. The increase in equivalent flat plate area for the heavy scout and heavy hog configurations increased nonlinearly at higher airspeeds. This nonlinear increase in equivalent flat plate area was probably caused by the change in aircraft attitude (nose down) as airspeed increased.

FIGURE A

CHANGE IN EQUIVALENT FLAT PLATE AREA
DUE TO WING ARMAMENT CONFIGURATION CHANGES

AH-IG USA S/N 615247
ROTOR SPEED= 324 RPM
DENSITY ALTITUDE=5000FT
GROSS WEIGHT=8500LB
C_T= 49.00 X 10^{-4}
C G LOCATION=FORWARD



15. The level flight and range performance summary for a thrust coefficient of 49.00 x 10^{-4} is presented in table 2. The value of 0.99 maximum nautical air miles per pound of fuel (NAMPP) decreased about 9.6 percent while the recommended cruise airspeed decreased 8.8 percent when comparing the minimum (clean) and maximum (heavy hog) aerodynamic drag configurations. The maximum airspeed in level flight decreased 8.4 percent in the maximum aerodynamic drag configuration.

16. The level flight performance presented in this report should be incorporated into the appropriate operator's manual.

Table 2. Level Flight and Range Performance Summary.

Standard day
Altitude - 5000 ft
Rotor speed - 324 rpm

Gross weight - 8500 lb
Center of gravity - fwd
No fairings on rocket pods

Configuration	Cruise Specific Range (NAMPP)	Recommend V_{Cruise} for 0.99 Maximum NAMPP (KTAS)	Maximum Airspeed in Level Flight (KTAS)
Clean	0.2270	138.0	149.0
Basic	0.2205	135.5	146.5
Light scout	0.2172	134.0	144.0
Inboard alternate	0.2162	133.0	143.5
Outboard alternate	0.2157	132.5	143.0
Heavy scout	0.2109	129.5	140.5
Heavy hog	0.2050	127.0	136.5

17. The production AH-1G aircraft equivalent flat plate area was increased approximately 5.0 square feet over that of the Bell Helicopter model 209 aircraft (ref 1, app I). The engine used during the evaluation of the Bell model 209 was not calibrated below an output torque pressure of 44.5 psi. Therefore, increase in equivalent flat plate area can only be calculated at engine output shaft horsepower above 1020. This increase in equivalent flat plate area was probably caused by these outside external changes:

a. The addition of two inboard wing stores stations.

b. The wider fuselage configuration for acceptance of the final chin turret.

c. The thicker stub wings.

d. Different configuration of the skid tubes and supporting structure.

e. The removal of flush-head rivets from the tail boom.

f. The addition of various access and vent panels.

AUTOROTATIONAL DESCENT PERFORMANCE

18. Steady state, autorotational descent performance tests were conducted in the unfaired heavy hog configuration at 5000 feet with an 8500-pound grwt and a forward cg. Results of the autorotational tests are presented in figure 15, appendix II.

19. The airspeed for minimum rate of descent (1815 ft/min) at these test conditions was 74.0 knots true airspeed (KTAS) at a rotor speed of 324 rpm. The rate of descent decreased as rotor speed was decreased from 324 rpm while maintaining a constant airspeed of 74.0 KTAS. The minimum rate of descent of 1750 ft/min was observed at the minimum rotor speed limit of 294 rpm. There was a definite increase in aircraft lateral vibration as rotor speed decreased to 310 rpm. The magnitude of these variations was not quantitatively measured. There was no noticeable decrease in aircraft response to a control input with decreasing rotor speed.

20. The autorotational characteristics of the AH-1G helicopter were found to be similar to the UH-1C helicopter. Precise control of rotor speed during autorotation was difficult because small collective control movements resulted in relatively large changes in rotor rpm. In addition, the high-inertia rotor system caused a lag in the response of rotor speed to collective control inputs. These two characteristics resulted in pilot tendency to "chase the rotor speed" (PRS A5). It was not difficult to maintain rotor speed between red lines, but maintainaing a selected rotor speed required considerable attention at a time when the pilot's attention should be directed outside the cockpit.

21. The autorotational descent performance presented should be incorporated into the appropriate operator's manual.

POWER AVAILABLE

22. All summary performance values were based upon shaft horsepower available as defined in figure 18, appendix II, since only the tail rotor drive shaft coupling was evaluated on the Phase B program. The power available charts presented were calculated by using the curves and calculation methods presented in model specification number 104.33 for the T53-L-13 engine (ref 4, app I).

23. In order to calculate shaft horsepower available, certain installation power losses had to be assumed or measured. Constant values were assumed for extracted shaft horsepower (zero) and power turbine output speed (6600 rpm). Power available was also calculated using zero, 0.6-percent and 3.6-percent engine bleed

air to consider the effects of one approved and one proposed aircraft modification. The approved aircraft modification uses engine bleed air to drive the engine oil cooler instead of the tail rotor drive shaft coupling. The proposed aircraft modification calls for the use of engine bleed air to drive a light-weight cockpit air conditioning system. The power available and fuel-flow characteristics are presented in figures 22 through 27, appendix II.

ENGINE INLET CHARACTERISTICS

24. The compressor inlet temperature and pressure recovery characteristics were considerably different for the production AH-1G than for the 209. Most of the change can be attributed to the production aircraft having an engine particle separator installed; the aircraft evaluated in reference 1, appendix I, was not so equipped. This change amounted to a decrease in engine power available of approximately 0.3 percent in a hover and 13.2 percent at 160 knots for a sea level, standard day for the production aircraft.

ENGINE CHARACTERISTICS

Static Stability

25. The engine static "droop" characteristics were good. Very few adjustments were required on the power turbine, speed-select "beep" switch when reducing or increasing engine power output. The engine power turbine speed-select "beep" switch characteristics are presented in figure 19, appendix II. The average time required for rotor speed to change after the "beep" switch was activated was 0.65 seconds. There was no noticeable variation in this delay time between a loaded or unloaded rotor system. The engine "beep" switch trim rate was constant at 7.7 rpm/sec after the time delay. The "beep" control characteristics were satisfactory and much improved over prior UH-1 series aircraft equipped with T53 series engines (PRS A3).

Dynamic Stability

26. Dymanic stability characteristics of the T53-L-13 engine appeared to be satisfactory throughout the flight envelope tested. When rapid power demands were required, no compressor stall was encountered during engine acceleration. Power overshoot and damping were satisfactory.

27. A slight engine oscillation was noted when operating the engine at maximum power available. This oscillation was not considered to be as serious as that reported in reference 10, appendix I. The horsepower fluctuation on the AH-1G was 10 to 15 shp. This fluctuation was not present when power was reduced to slightly below the maximum available.

28. There was only one significant airframe/engine matching shortcoming discovered during this program. A problem was encountered in forward flight when an abrupt maneuver requiring left-lateral cyclic was initiated with the aircraft operating at or near the main-transmission torque limit. The abrupt left-lateral cyclic input caused the rotor speed to decrease while maintaining a constant collective control position. The engine power-turbine governor sensed the rotor speed decrease and increased the fuel flow. This resulted in an increased engine power output and a main transmission overtorque condition. This characteristic is transient in nature and was only encountered in forward flight conditions below the critical altitude of the engine. The largest change in engine torque observed during the program was 13 psi for a left-lateral control input of 3.3 inches at a trim calibrated airspeed of 105 knots. In order to avoid overtorquing of the main transmission, the pilot must continually monitor the torquemeter when performing an abrupt left-lateral cyclic control input. This characteristic is undesirable since the attention of the pilot may be required elsewhere when performing a mission. This condition detracts from the mission capability of the aircraft. Abrupt right cyclic inputs under the same conditions have just the opposite effect: an increase in rotor speed and decrease in engine power output. This problem has been reported previously in reference 2, appendix I.

DIRECTIONAL CONTROL EVALUATION

29. An IGE directional control evaluation was conducted with the canopy doors on and off to determine if there was any significant change in handling qualities of the aircraft. The test was conducted with the SCAS yaw channel OFF and in the outboard alternate configuration. The ground paced method of test was used with conditions limited to an airspeed range from zero to 17.5 KTAS at critical wind azimuths of 160, 200 and 240 degrees (clockwise from nose of aircraft). The results of these tests are graphically presented in figures 16 through 18, appendix II.

30. The test revealed little if any change in the aircraft directional handling qualities with the canopy doors on or off. However, an area of directional instability existed between 9 and 13

knots at each wind azimuth flown. In this area, rapid and sometimes large directional control movements were required to maintain the desired heading. This instability is a major deficiency and detracts from the mission suitability of the aircraft (PRS A6).

31. The directional control evaluation reported in reference 8, appendix I, was found to agree with the data presented in this report. Some scatter in the data was noted where directional instability occurred but was not considered to be significant.

AIRSPEED CALIBRATION

32. The helicopter was equipped with a test airspeed indicator system (boom) in addition to the standard helicopter airspeed indicator system. Airspeed calibration flights were conducted to determine the position error of the test system. A trailing bomb was used as an airspeed reference up to 101 knots. From 80 to 180 knots indicated airspeed (KIAS), a T-28 airplane with a calibrated airspeed system was used as an airspeed reference. The test aircraft system was also calibrated (between 39 and 159 KTAS) using the ground speed course. Calibration of the test system was conducted in the clean configuration in level flight, dive, climb and autorotation at a rotor speed of 324 rpm. The test results are presented in figure 28, appendix II.

CONCLUSIONS

33. The following conclusions were reached after completion of the AH-1G Phase B, Part 6 performance tests:

a. The equivalent flat plate area can increase as much as 7.7 square feet depending on wing armament configuration. This increase in equivalent flat plate area decreased the specific range of the aircraft 9.6 percent (para 15).

b. Changes in the production fuselage increased the equivalent flat plate area by 5.0 square feet over the aircraft reported in reference 1, appendix I (para 17).

c. The minimum steady state autorotational descent in the heavy hog configuration is 1815 ft/min for a rotor speed of 324 rpm at a true airspeed of 74.0 knots (para 19).

d. The steady state autorotational rate of descent decreased to 1750 ft/min for a rotor speed of 294 rpm at a true airspeed of 74.0 knots (para 19).

e. Precise control of rotor speed during autorotation was difficult because small collective control movements resulted in relatively large changes in rotor speed (para 20).

f. The changes in the production inlet configuration increased the engine power available loss as much as 13.2 percent depending on airspeed (para 24).

g. The removal of the canopy door did not significantly affect the low-speed directional control margin or IGE flying qualities (para 30).

34. Correction of the following deficiency is mandatory for acceptance of the aircraft: Directional instability existed between 9 and 13 knots for the wind azimuths between 160 and 240 degrees (para 30).

35. Correction of the following shortcoming is desirable for acceptance of the aircraft weapons system: Undue pilot attention required to avoid overtorquing the main transmission during maneuvers requiring abrupt left-lateral cyclic inputs (para 28).

RECOMMENDATIONS

36. The performance data presented in this report should be incorporated into the operator's manual (paras 16 and 21).

37. The shortcomings should be corrected on a high-priority basis (para 30).

APPENDIX I. REFERENCES

1. Final Report, USAAVNTA, Project No. 65-30, *Engineering Flight Evaluation of the Bell Model 209 Armed Helicopter*, May 1966.

2. Final Report, USAAVNTA, Project No. 66-06, *Engineering Flight Test of the AH-1G Helicopter, Hueycobra, Phase B, Part 1*, January 1968.

3. Specification, 209-947-530, Bell Helicopter Company, *Detail Specification for Model AH-1G Helicopter*, 25 September 1967, revised 29 December 1967.

4. Specification, 104.33, Lycoming Division of Avco Corporation, *Model Specification T53-L-13 Shaft Turbine Engine*, 6 May 1966.

5. Letter, STEAP-DS-TI, Aberdeen Proving Ground, subject: Test Directive, Engineering/Logistical Evaluation Test of AH-1G Helicopter (Hueycobra), 13 September 1966.

6. Plan of Test, USAAVNTA, Project No. 66-06, *Engineering Flight Test of the AH-1G Helicopter (Hueycobra) Phase B*, April 1967.

7. Message, USAAVSCOM, AMSAV-EF, No. 1367, Unclas, subject: AH-1G Phase B Performance Testing and Instrumentation, 17 August 1967.

8. Final Report, USAAVNTA, Project No. 66-06, *Engineering Flight Test of the AH-1G Helicopter to Determine the Area of Inadequate Directional Control Power at 8100 Pounds Gross Weight*, February 1968.

9. Final Report, USAAVNTA, Project No. 66-06, *Engineering Flight Evaluation of the AH-1G Helicopter, Hueycobra, Phase B, Part 2*, February 1969.

10. Final Report, USAAVNTA, Project No. 66-04, *Engineering Flight Test of the UH-1H Helicopter, Phase D Product Improvement Test*, August 1967.

APPENDIX II. TEST DATA

FIGURE No 1

CHANGE IN EQUIVALENT FLAT PLATE AREA DUE TO WING ARMAMENT CONFIGURATION CHANGES

AH-1G USA S/N 615247

ROTOR SPEED = 324 RPM
DENSITY ALTITUDE = 5000 FT.
GROSS WEIGHT = 8500 LB
$C_T = 49.00 \times 10^{-4}$
C.G. LOCATION = FORWARD

NOTE: ALL ROCKET PODS UNFAIRED

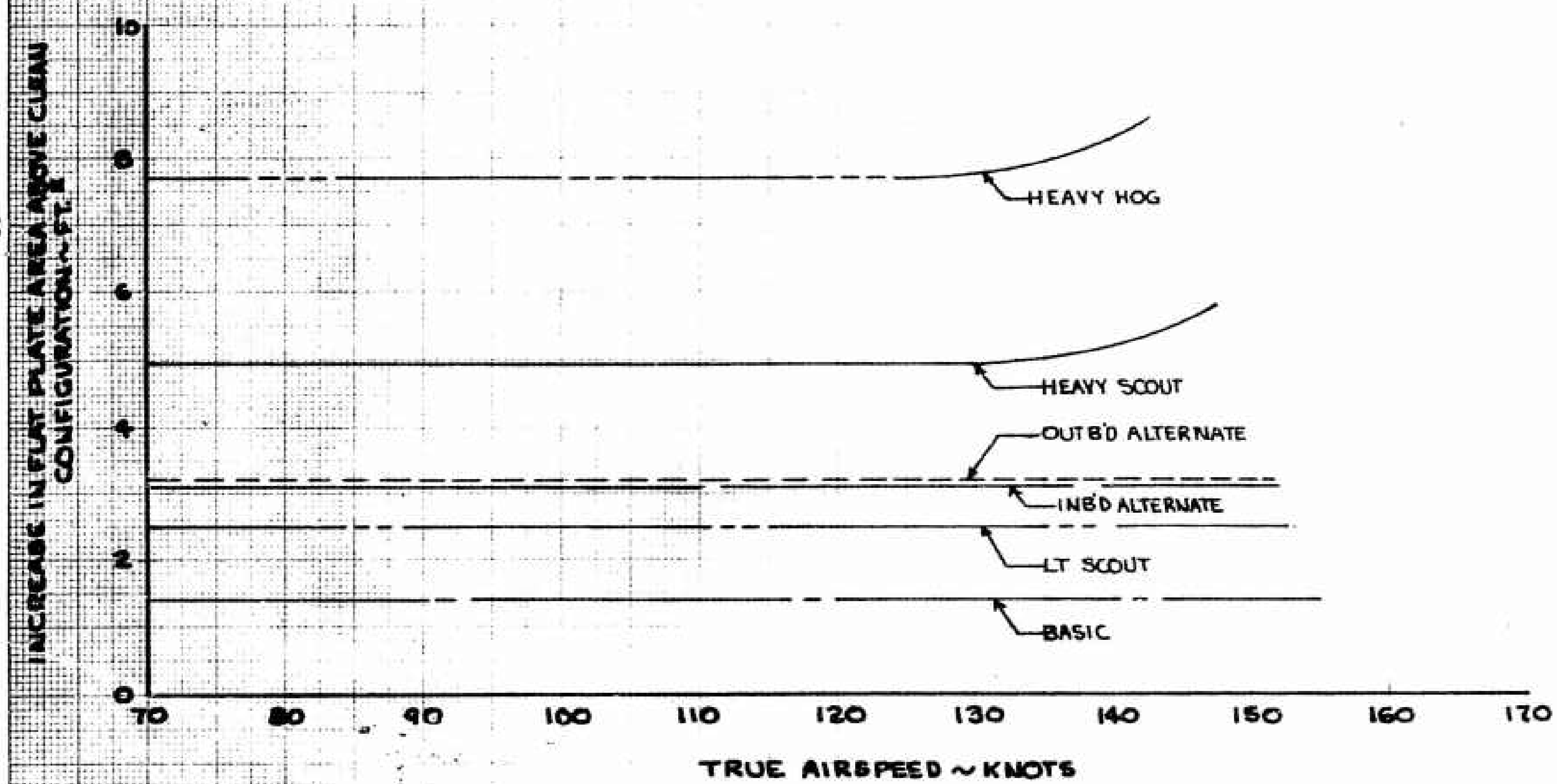


FIGURE NO. 2

NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

HEAVY HOG CONFIGURATION

(UNFAIRED)

CENTER OF GRAVITY-FORWARD

NOTE: POINTS OBTAINED FROM FIGURE 4 THROUGH 8 APPENDIX II

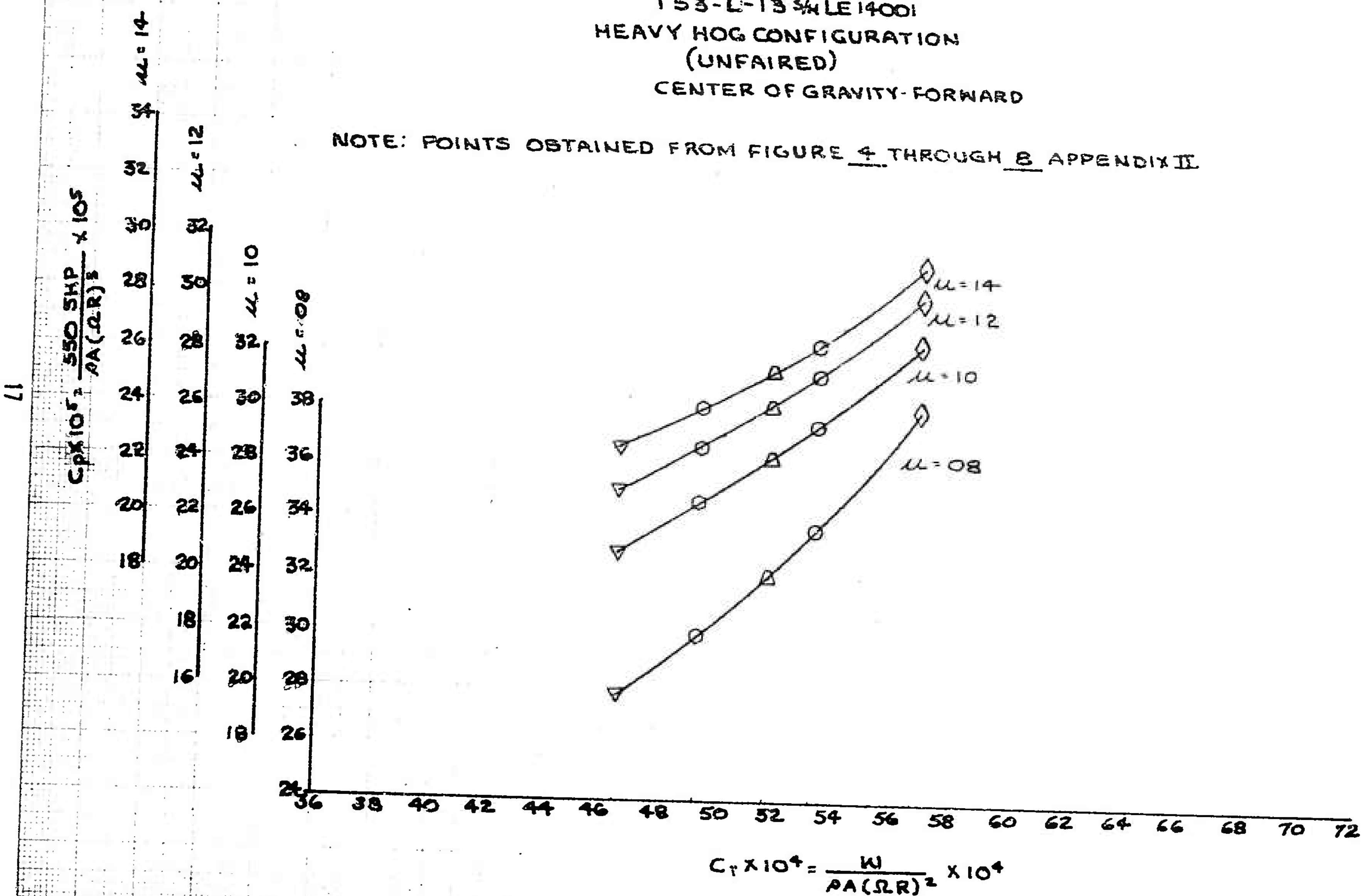


FIGURE NO. 3

NON-DIMENSIONAL LEVEL FLIGHT PERFORMANCE

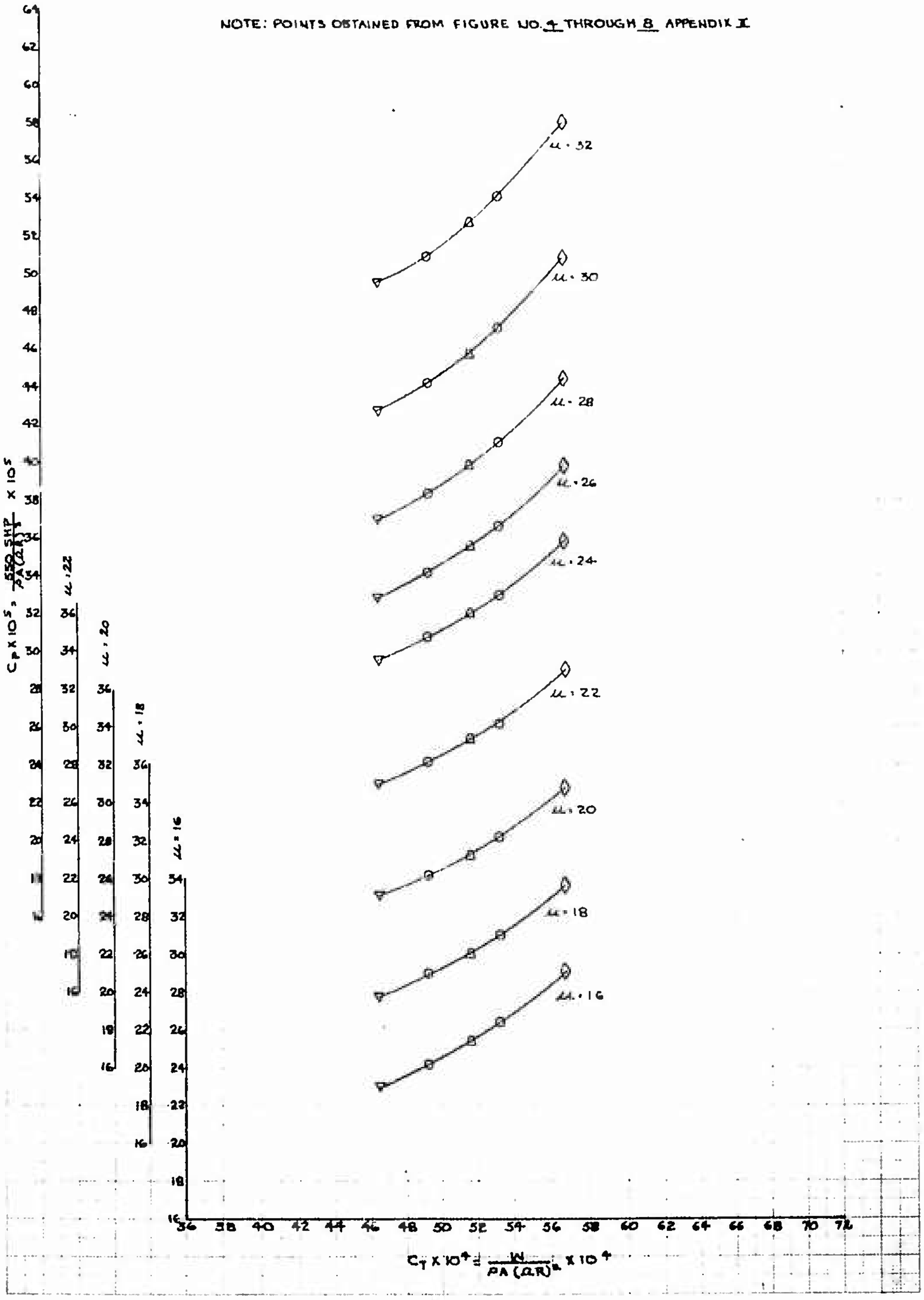
AH-1G USA S/N 615247

T53-L-13 S/N LE 14001

HEAVY HOG CONFIGURATION

(UNFAIRED)

CENTER OF GRAVITY-FORWARD



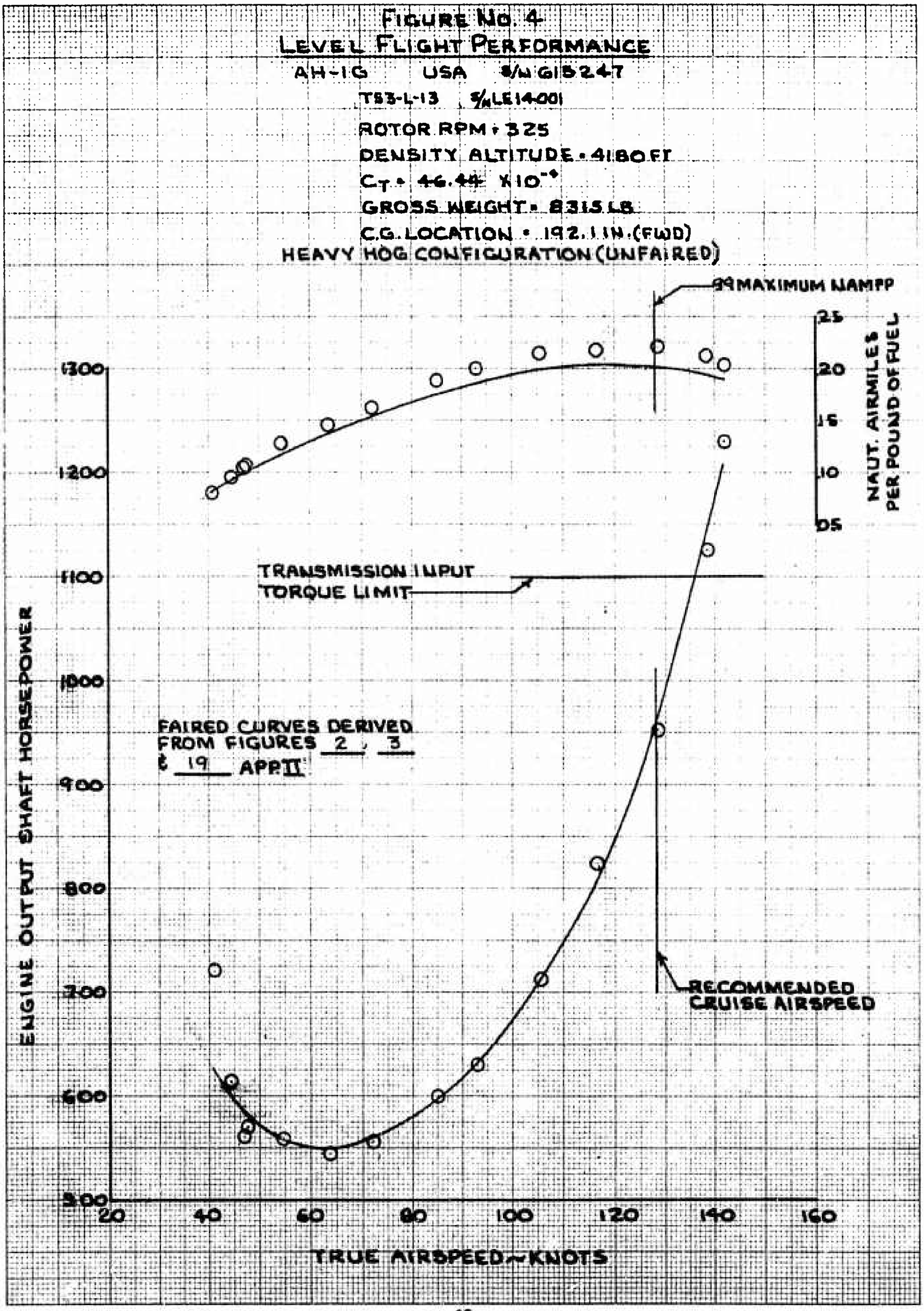


FIGURE NO. 4
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 325
DENSITY ALTITUDE = 4180 FT
C_T = 46.44 X 10^{-4}
GROSS WEIGHT = 8315 LB
C.G. LOCATION = 192.1 IN. (FWD)
HEAVY HOG CONFIGURATION (UNFAIRED)
99 MAXIMUM NAMPP
NAUT. AIRMILES PER POUND OF FUEL
TRANSMISSION INPUT TORQUE LIMIT
FAIRED CURVES DERIVED FROM FIGURES 2, 3 & 19 APP II
RECOMMENDED CRUISE AIRSPEED
ENGINE OUTPUT SHAFT HORSEPOWER
TRUE AIRSPEED ~ KNOTS

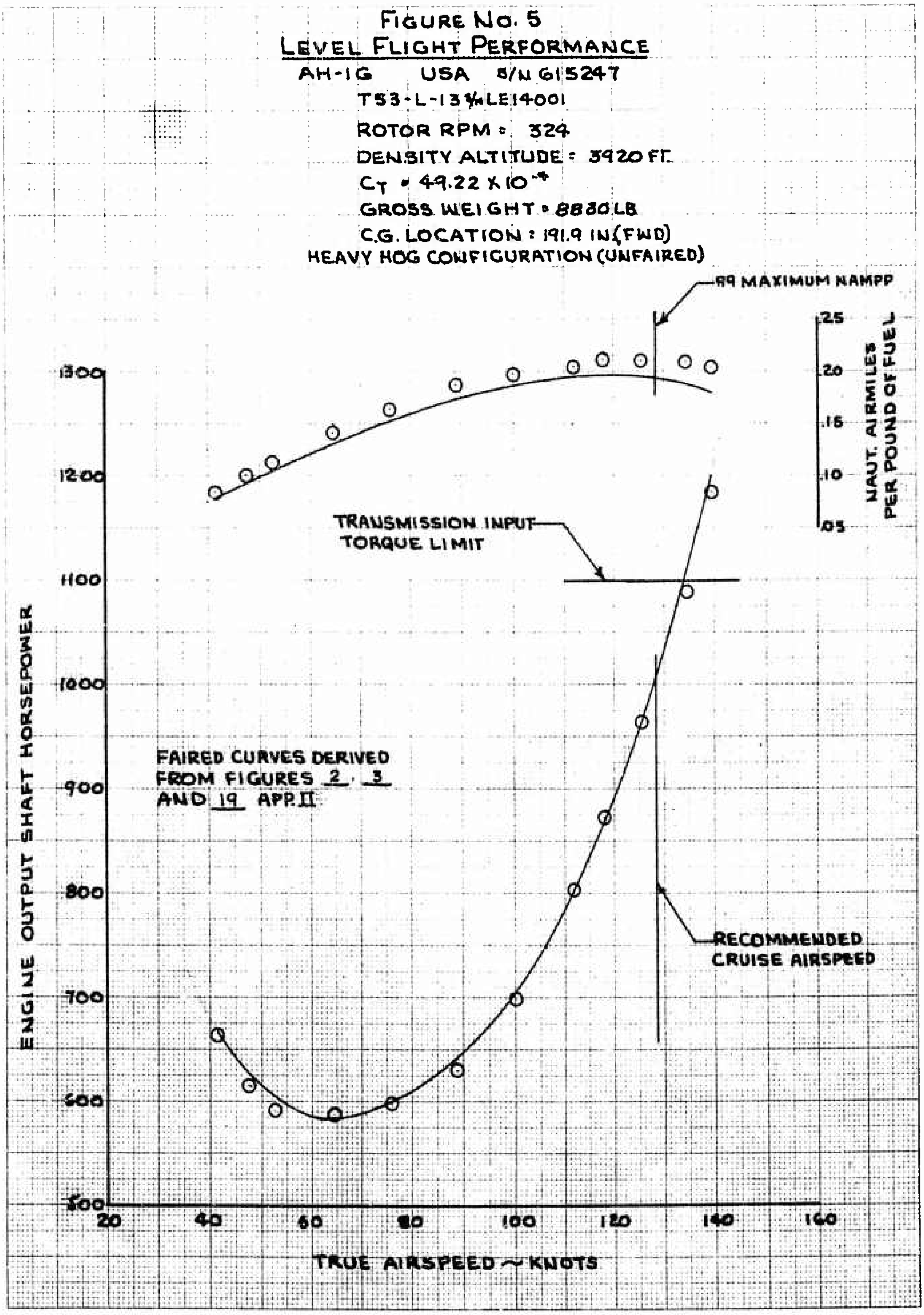


FIGURE NO. 5
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 324
DENSITY ALTITUDE = 3920 FT.
CT = 49.22 X 10^-4
GROSS WEIGHT = 8830 LB
C.G. LOCATION = 191.9 IN.(FWD)
HEAVY HOG CONFIGURATION (UNFAIRED)
.99 MAXIMUM NAMPP
NAUT. AIRMILES PER POUND OF FUEL
TRANSMISSION INPUT TORQUE LIMIT
FAIRED CURVES DERIVED FROM FIGURES 2, 3 AND 19 APP.II
RECOMMENDED CRUISE AIRSPEED
ENGINE OUTPUT SHAFT HORSEPOWER
TRUE AIRSPEED ~ KNOTS

FIGURE NO. 6
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE 14001
ROTOR RPM = 324.5
DENSITY ALTITUDE = 4190 FT.
C_T = 51.63 X 10^{-6}
GROSS WEIGHT = 9210 LB.
C.G. LOCATION = 193.0 IN. (FWD)
HEAVY HOG CONFIGURATION (UNFAIRED)

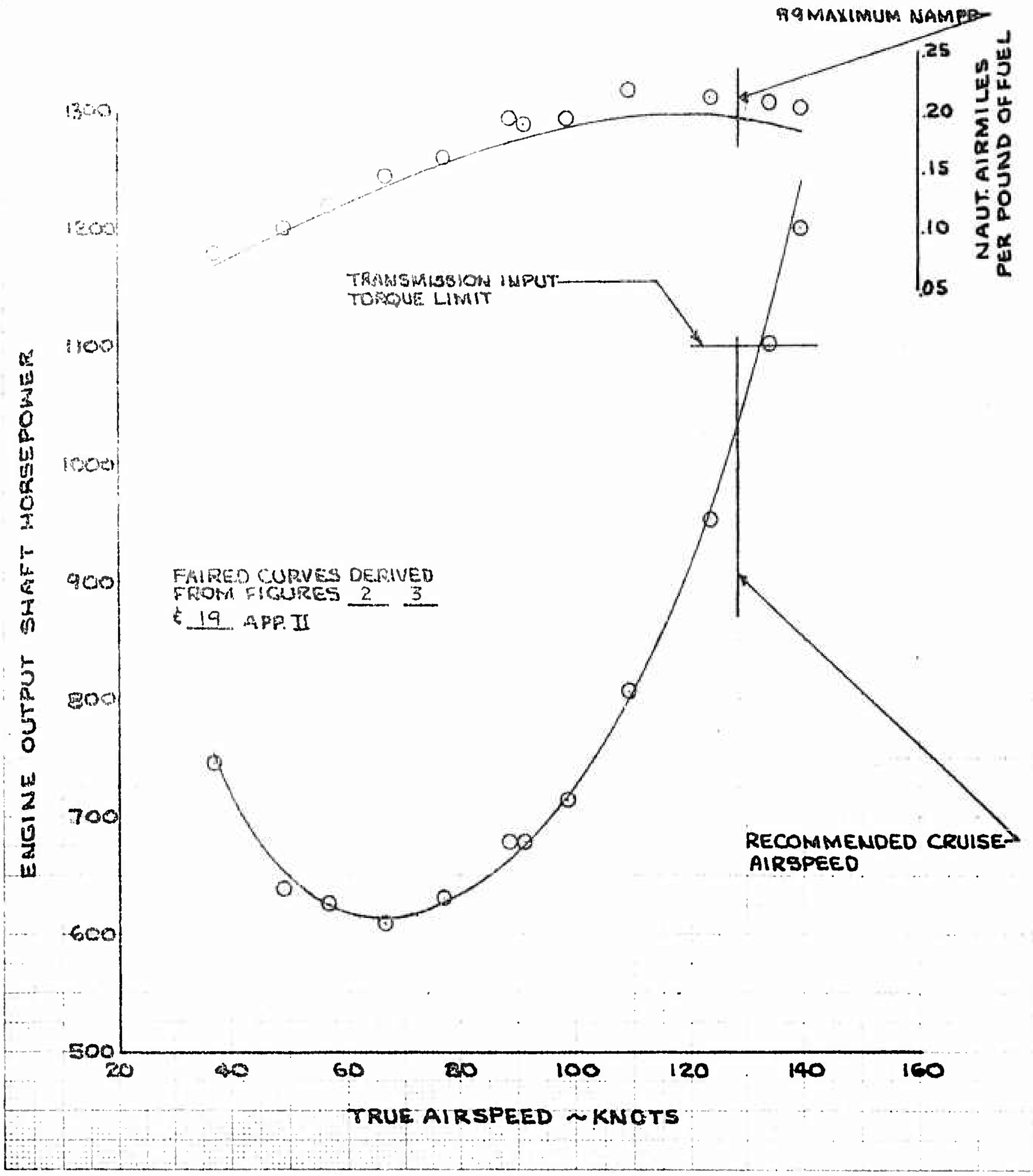


FIGURE No. 7

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615 247

T53-L-13 S/N LE14001

ROTOR RPM = 324.5

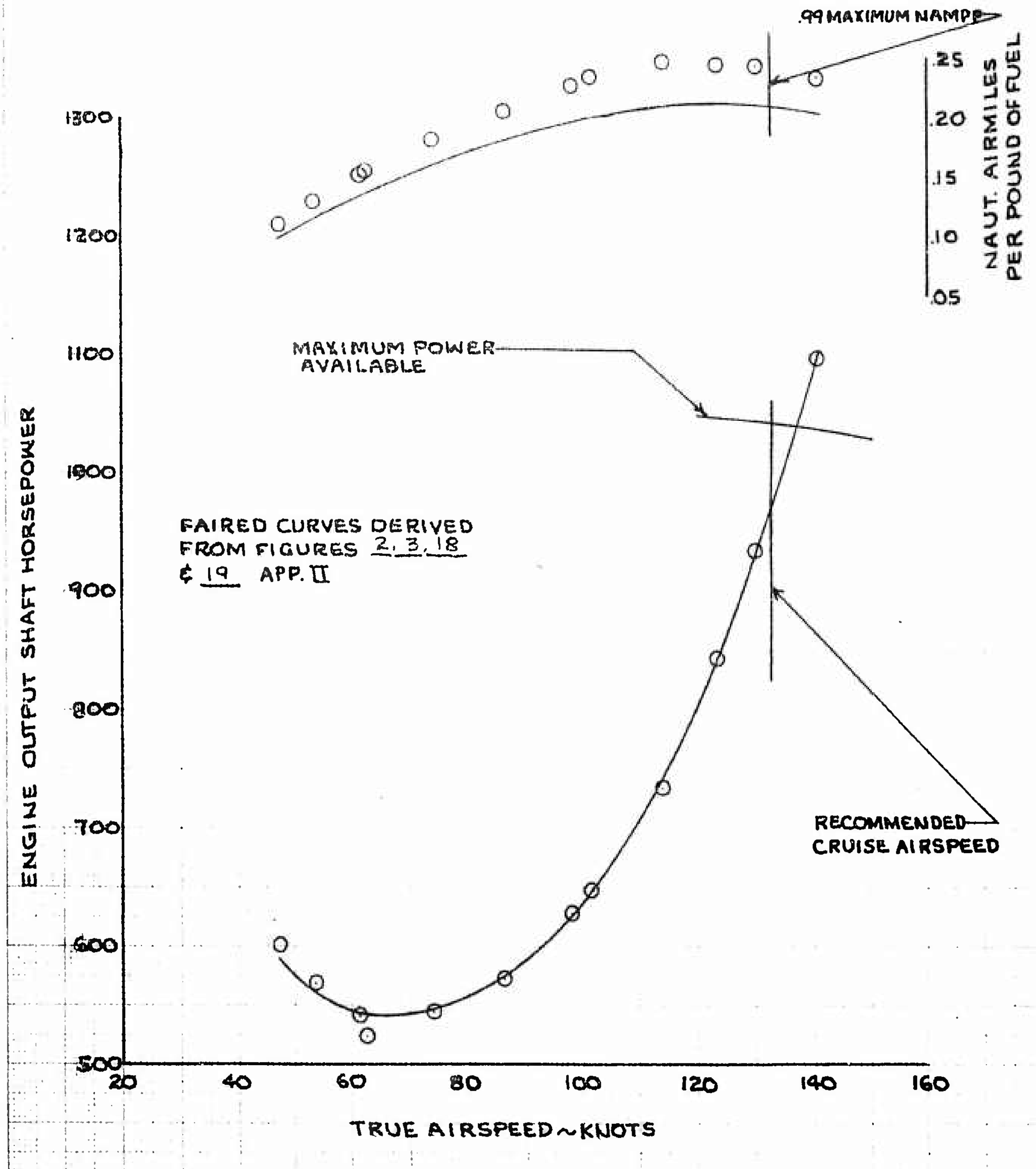
DENSITY ALTITUDE = 9470 FT.

C_T = 53.22 X10^{-4}

GROSS WEIGHT = 8075 LB

C.G. LOCATION = 192.6 IN.(FWD

HEAVY HOG CONFIGURATION (UNFAIRED)



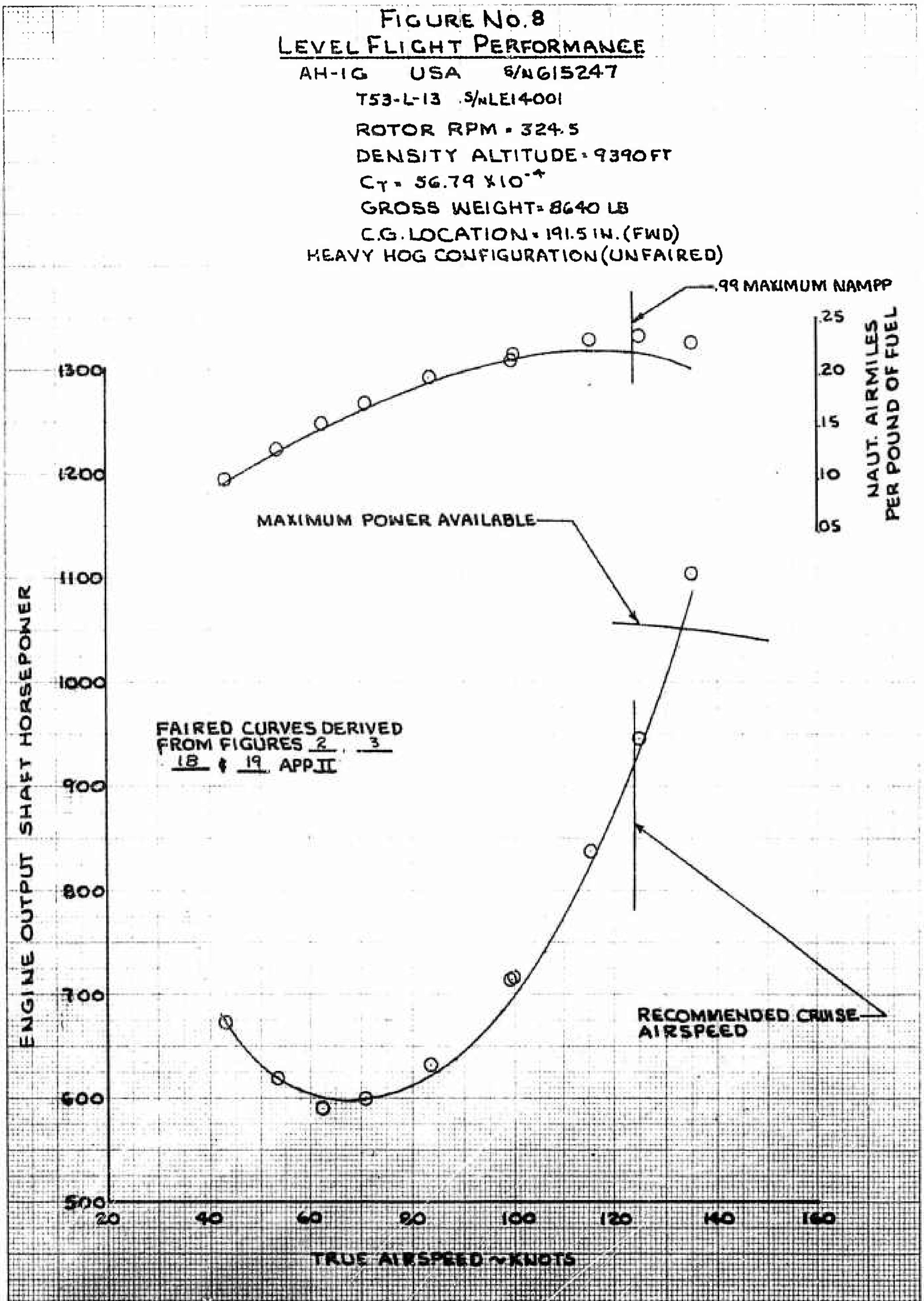


FIGURE No. 8
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 324.5
DENSITY ALTITUDE = 9390 FT
C_T = 56.79 X 10^{-4}
GROSS WEIGHT = 8640 LB
C.G. LOCATION = 191.5 IN. (FWD)
HEAVY HOG CONFIGURATION (UNFAIRED)
.99 MAXIMUM NAMPP
MAXIMUM POWER AVAILABLE
FAIRED CURVES DERIVED FROM FIGURES 2, 3, 18 & 19 APP II
RECOMMENDED CRUISE AIRSPEED
ENGINE OUTPUT SHAFT HORSEPOWER
NAUT. AIRMILES PER POUND OF FUEL
TRUE AIRSPEED ~ KNOTS
500
600
700
800
900
1000
1100
1200
1300
.05
.10
.15
.20
.25
20
40
60
80
100
120
140
160

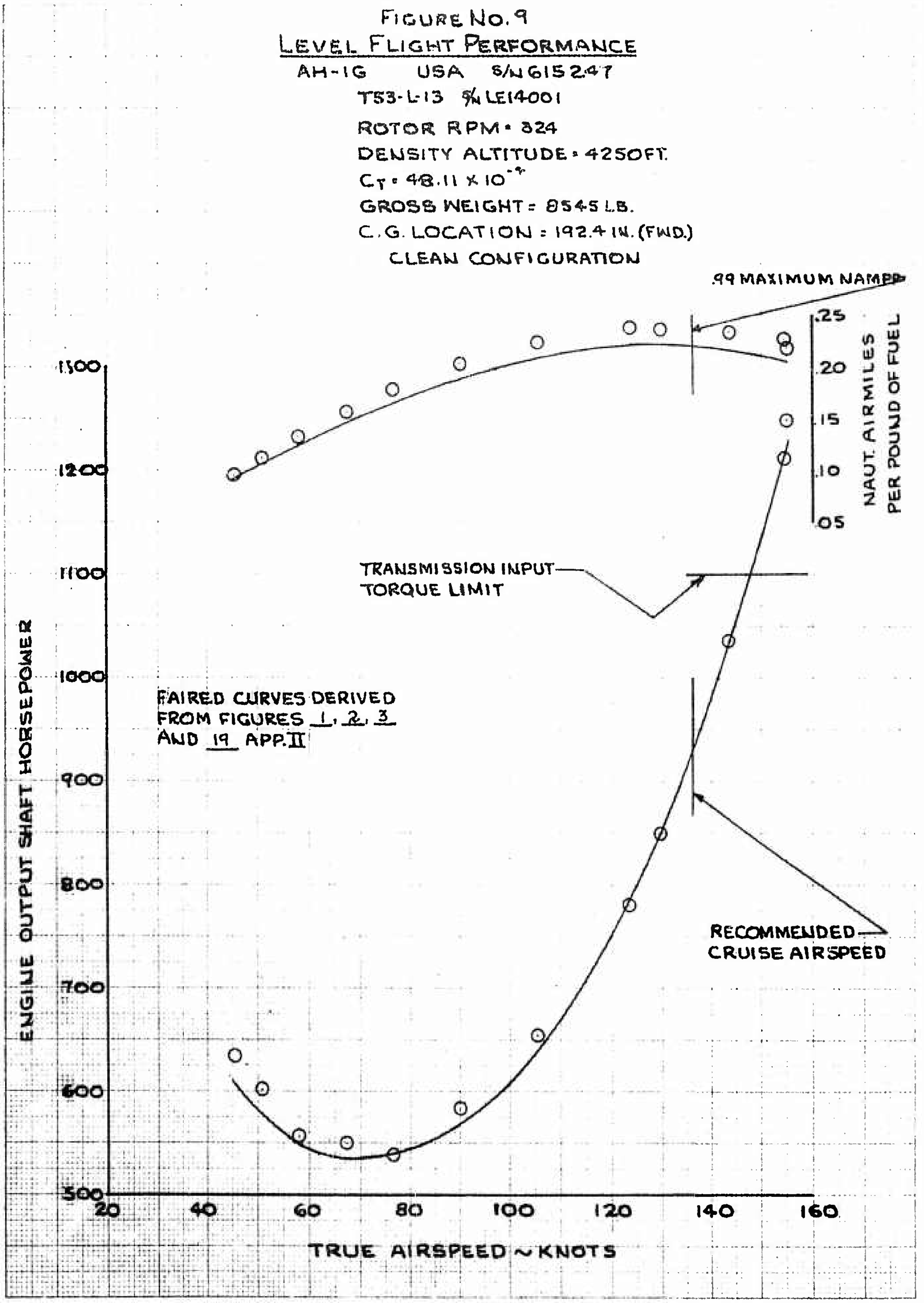


FIGURE No. 9
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 324
DENSITY ALTITUDE = 4250FT.
CT = 48.11 x 10^-4
GROSS WEIGHT = 8545 LB.
C.G. LOCATION = 192.4 IN. (FWD.)
CLEAN CONFIGURATION
.99 MAXIMUM NAMPP
TRANSMISSION INPUT TORQUE LIMIT
FAIRED CURVES DERIVED FROM FIGURES 1, 2, 3 AND 19 APP. II
RECOMMENDED CRUISE AIRSPEED
ENGINE OUTPUT SHAFT HORSEPOWER
NAUT. AIRMILES PER POUND OF FUEL
TRUE AIRSPEED ~ KNOTS

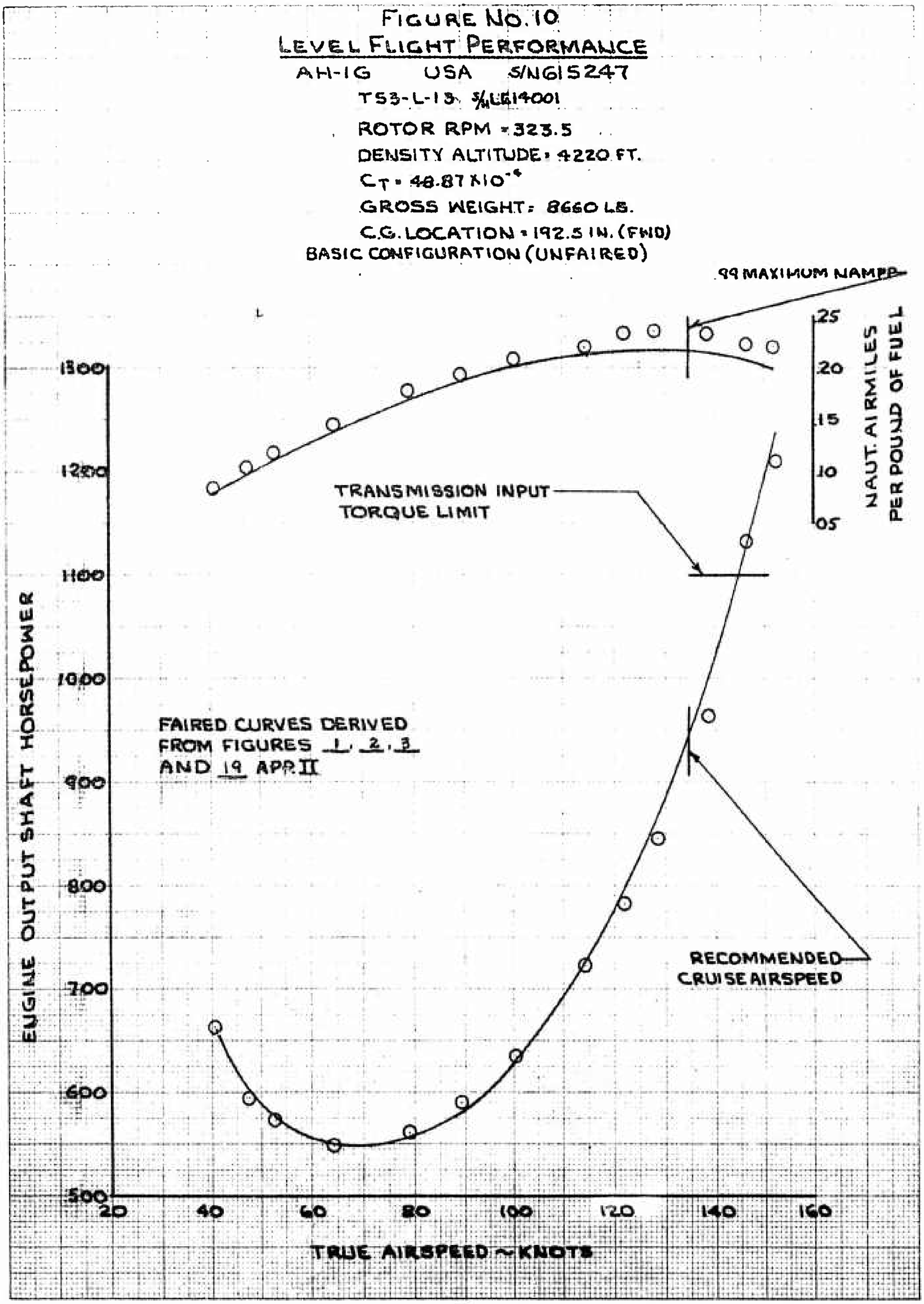


FIGURE NO. 10
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 323.5
DENSITY ALTITUDE = 4220 FT.
C_T = 48.87 X 10^{-4}
GROSS WEIGHT = 8660 LB.
C.G. LOCATION = 192.5 IN. (FWD)
BASIC CONFIGURATION (UNFAIRED)
.99 MAXIMUM NAMPP
NAUT. AIRMILES PER POUND OF FUEL
TRANSMISSION INPUT TORQUE LIMIT
FAIRED CURVES DERIVED FROM FIGURES 1, 2, 3 AND 19 APP. II
RECOMMENDED CRUISE AIRSPEED
ENGINE OUTPUT SHAFT HORSEPOWER
TRUE AIRSPEED ~ KNOTS
1300
1200
1100
1000
900
800
700
600
500
20
40
60
80
100
120
140
160
.25
.20
.15
.10
.05

FIGURE No. 11

LEVEL FLIGHT PERFORMANCE

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

ROTOR RPM = 325.5

DENSITY ALTITUDE = 4300 FT.

C_T = 48.30 × 10^{-4}

GROSS WEIGHT = 8645 LB

C.G. LOCATION = 192.2 IN (FWD)

LIGHT SCOUT CONFIGURATION (UNFAIRED)

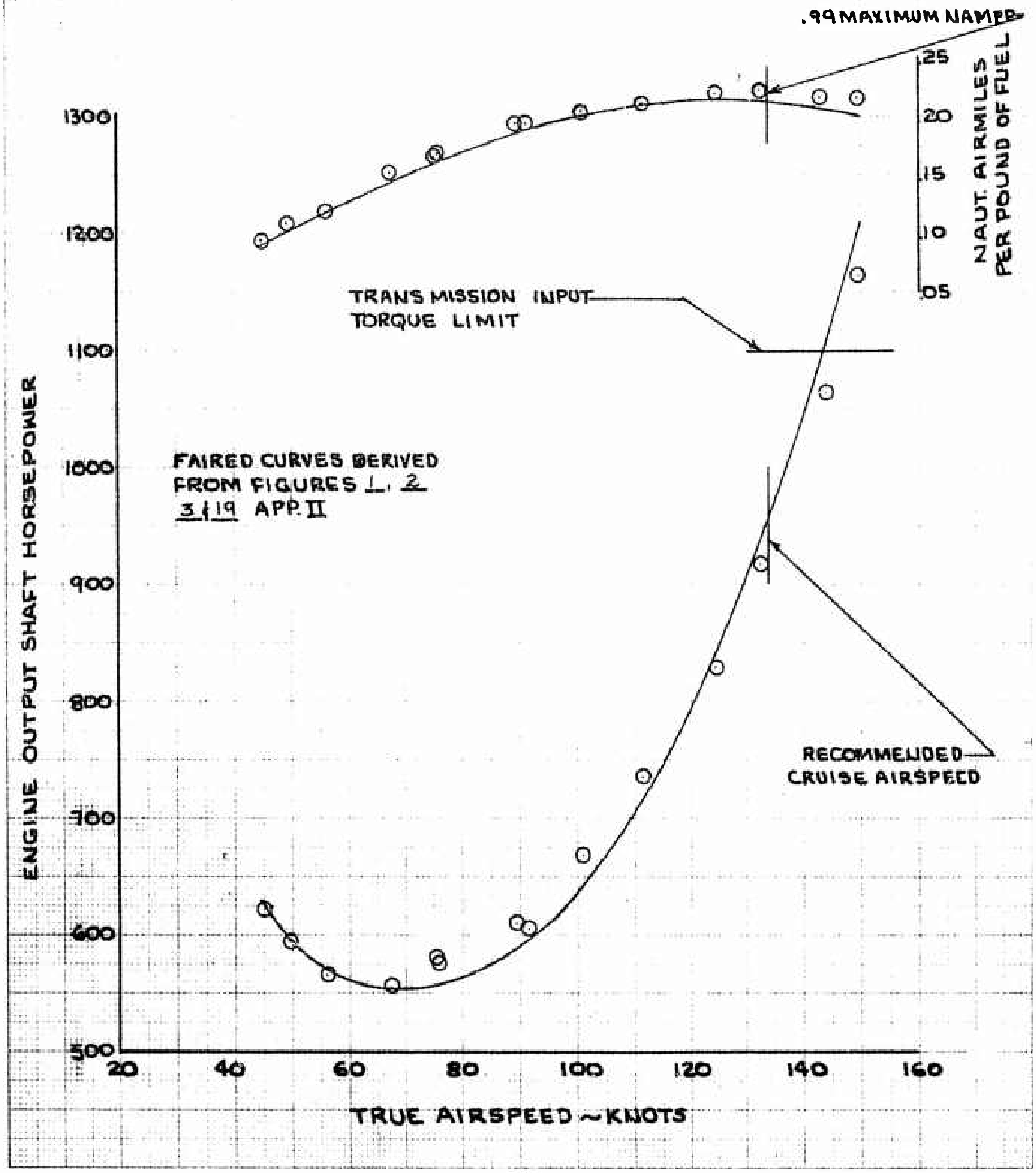


FIGURE No 12
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 323.5
DENSITY ALTITUDE = 4120 FT.
C_T = 49.32 × 10^{-4}
GROSS WEIGHT = 8765 LB.
C.G. LOCATION = 192.8 IN. (FWD)
INBOARD ALTERNATE CONFIGURATION (UNFAIRED)

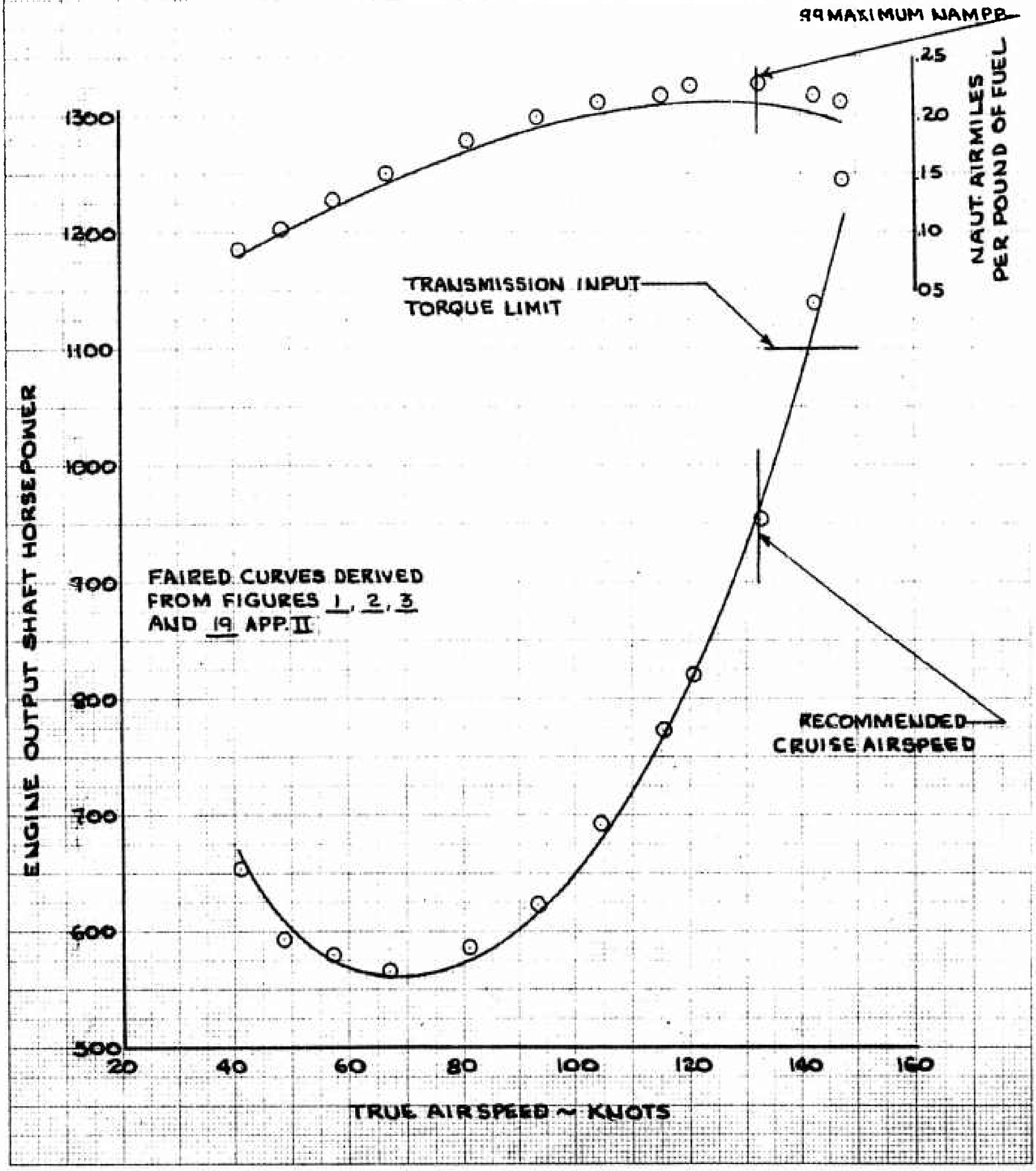
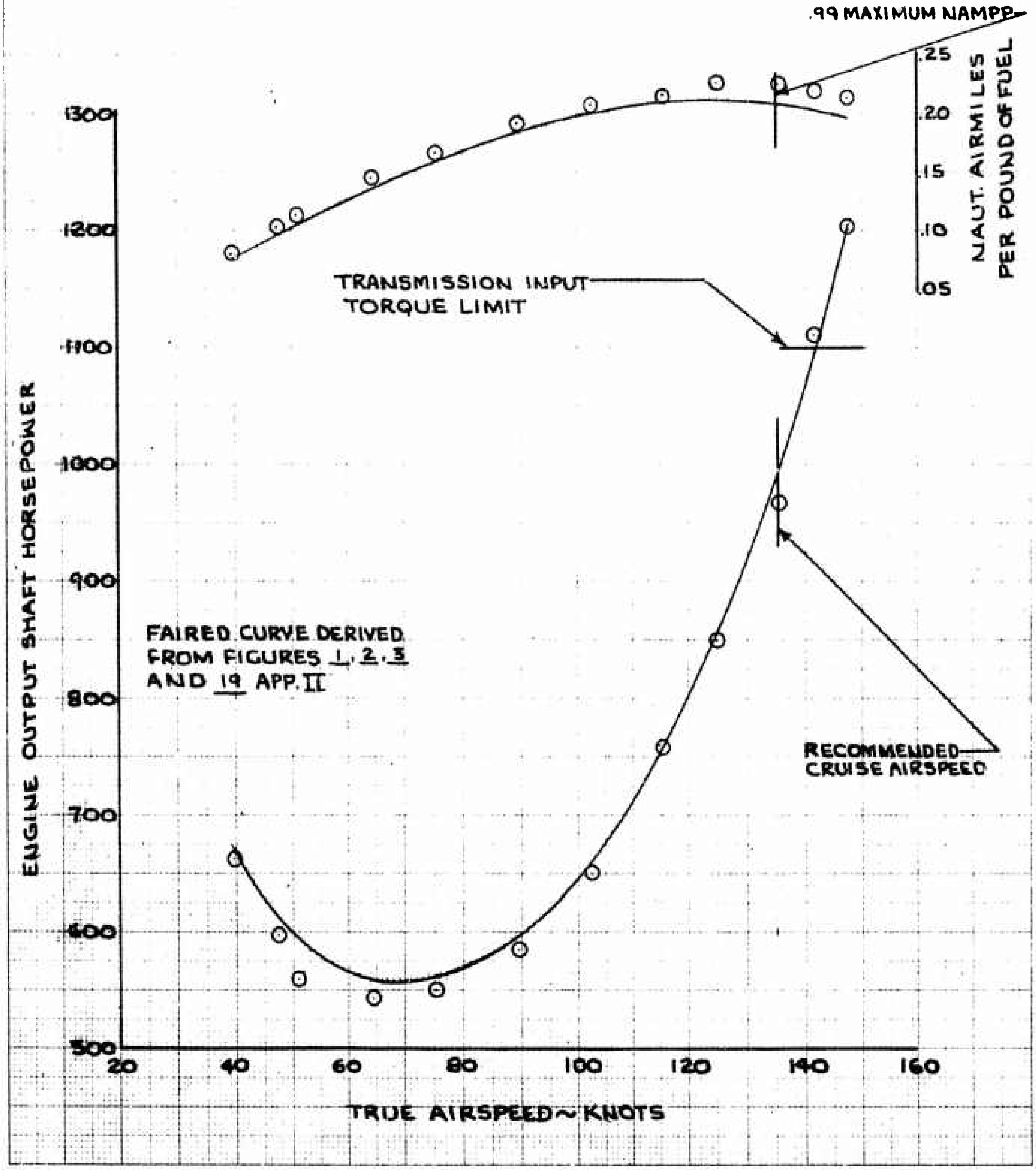


FIGURE No. 13
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 323.5
DENSITY ALTITUDE = 3960 FT
$C_T = 48.81 \times 10^{-4}$
GROSS WEIGHT = 8715 LBS
C.G. LOCATION = 193.0 IN. (FWD)
OUTBOARD ALTERNATE CONFIGURATION (UNFAIRED)



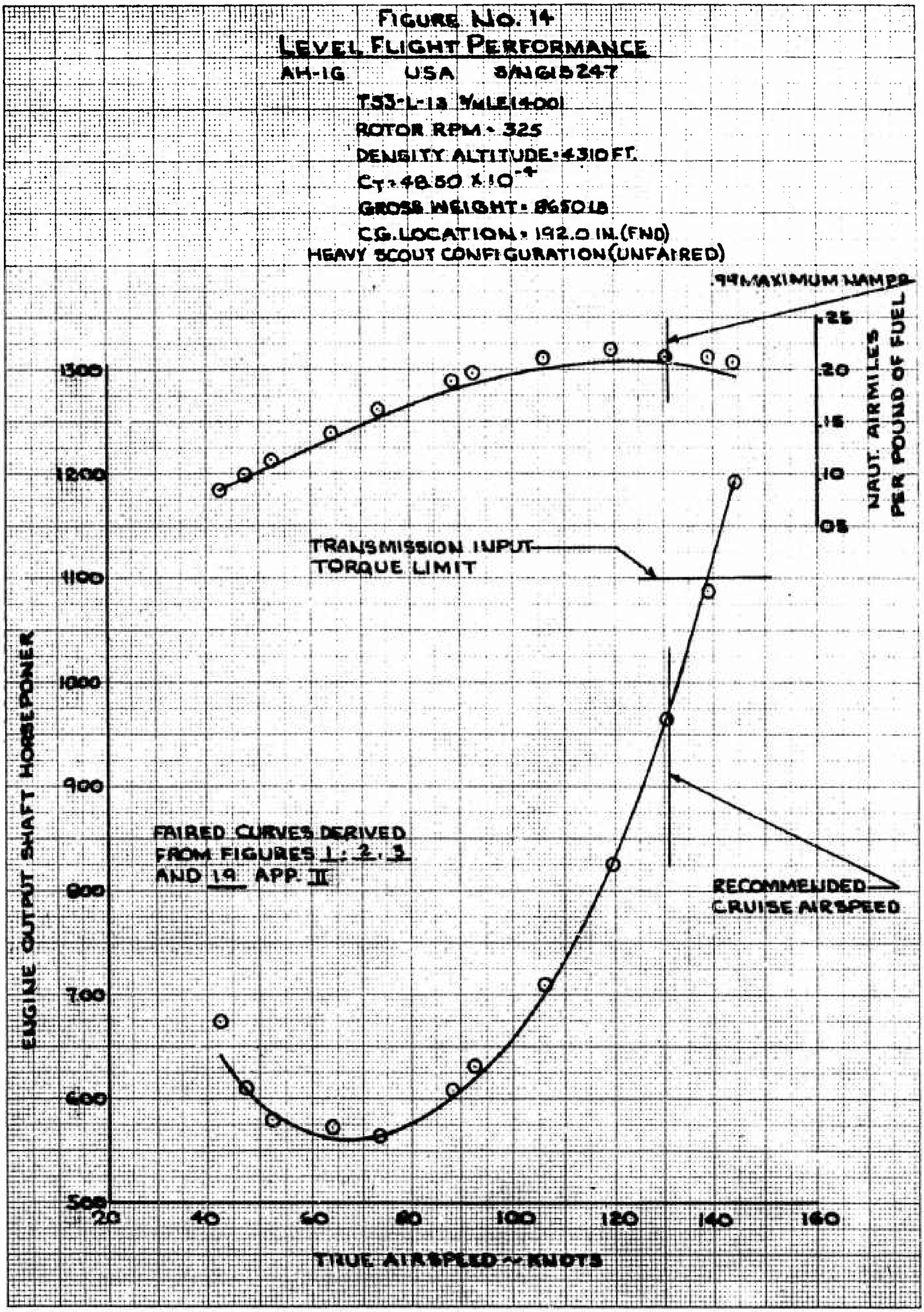


FIGURE NO. 14
LEVEL FLIGHT PERFORMANCE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ROTOR RPM = 325
DENSITY ALTITUDE = 4310 FT.
C_T = 48.50 × 10^{-4}
GROSS WEIGHT = 8650 LB
C.G. LOCATION = 192.0 IN. (FWD)
HEAVY SCOUT CONFIGURATION (UNFAIRED)
99% MAXIMUM NAMPP
NAUT. AIRMILES PER POUND OF FUEL
.25
.20
.15
.10
.05
TRANSMISSION INPUT TORQUE LIMIT
FAIRED CURVES DERIVED FROM FIGURES 1, 2, 3 AND 19 APP. II
RECOMMENDED CRUISE AIRSPEED
ENGINE OUTPUT SHAFT HORSEPOWER
1300
1200
1100
1000
900
800
700
600
500
20
40
60
80
100
120
140
160
TRUE AIRSPEED ~ KNOTS

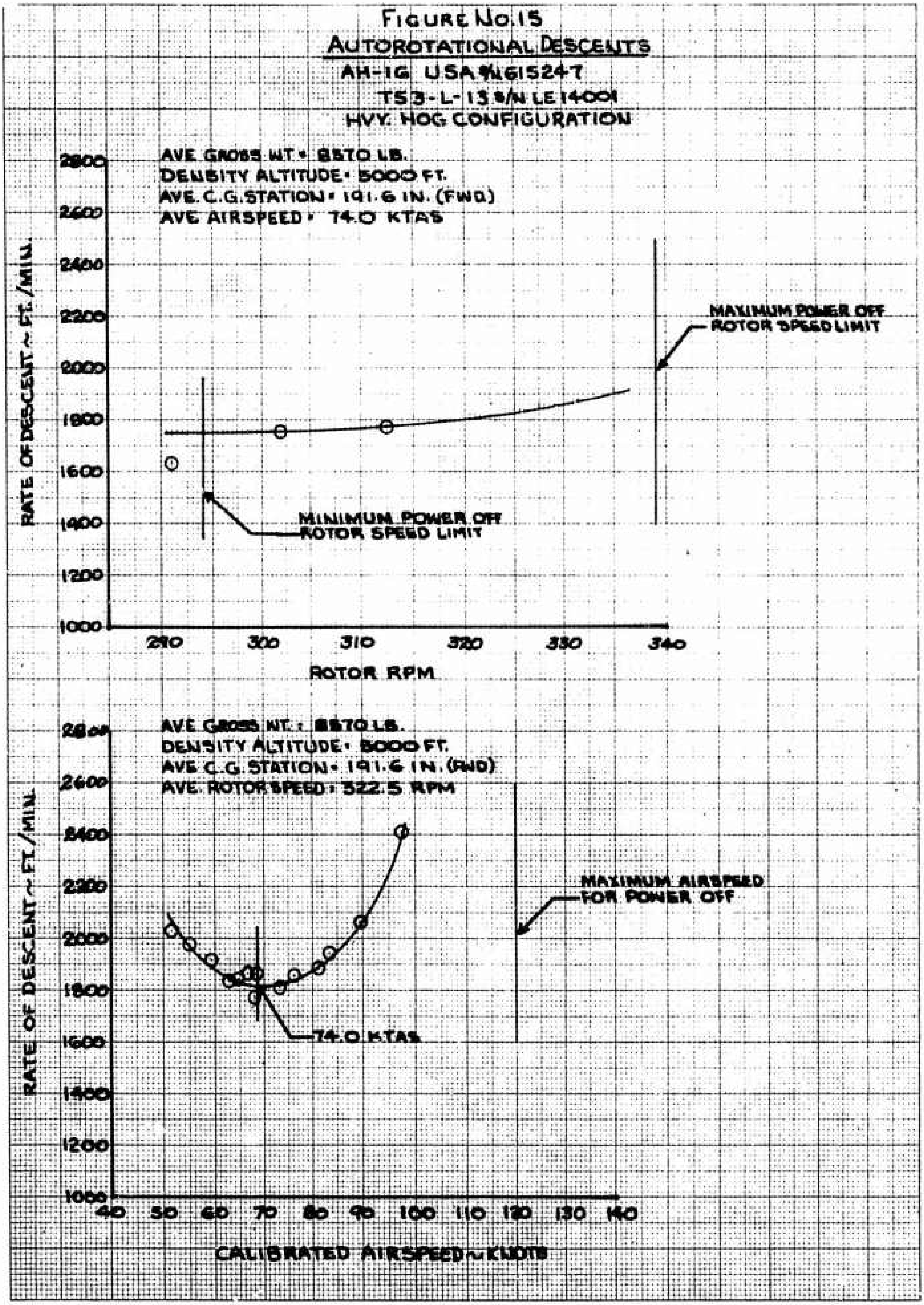


FIGURE NO. 15
AUTOROTATIONAL DESCENTS
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
HVY. HOG CONFIGURATION
AVE GROSS WT = 8570 LB.
DENSITY ALTITUDE = 5000 FT.
AVE. C.G. STATION = 191.6 IN. (FWD)
AVE AIRSPEED = 74.0 KTAS
MAXIMUM POWER OFF ROTOR SPEED LIMIT
MINIMUM POWER OFF ROTOR SPEED LIMIT
RATE OF DESCENT ~ FT./MIN.
ROTOR RPM
AVE GROSS WT. = 8570 LB.
DENSITY ALTITUDE = 5000 FT.
AVE C.G. STATION = 191.6 IN. (FWD)
AVE. ROTOR SPEED = 322.5 RPM
MAXIMUM AIRSPEED FOR POWER OFF
74.0 KTAS
RATE OF DESCENT ~ FT./MIN.
CALIBRATED AIRSPEED ~ KNOTS

FIGURE NO. 16

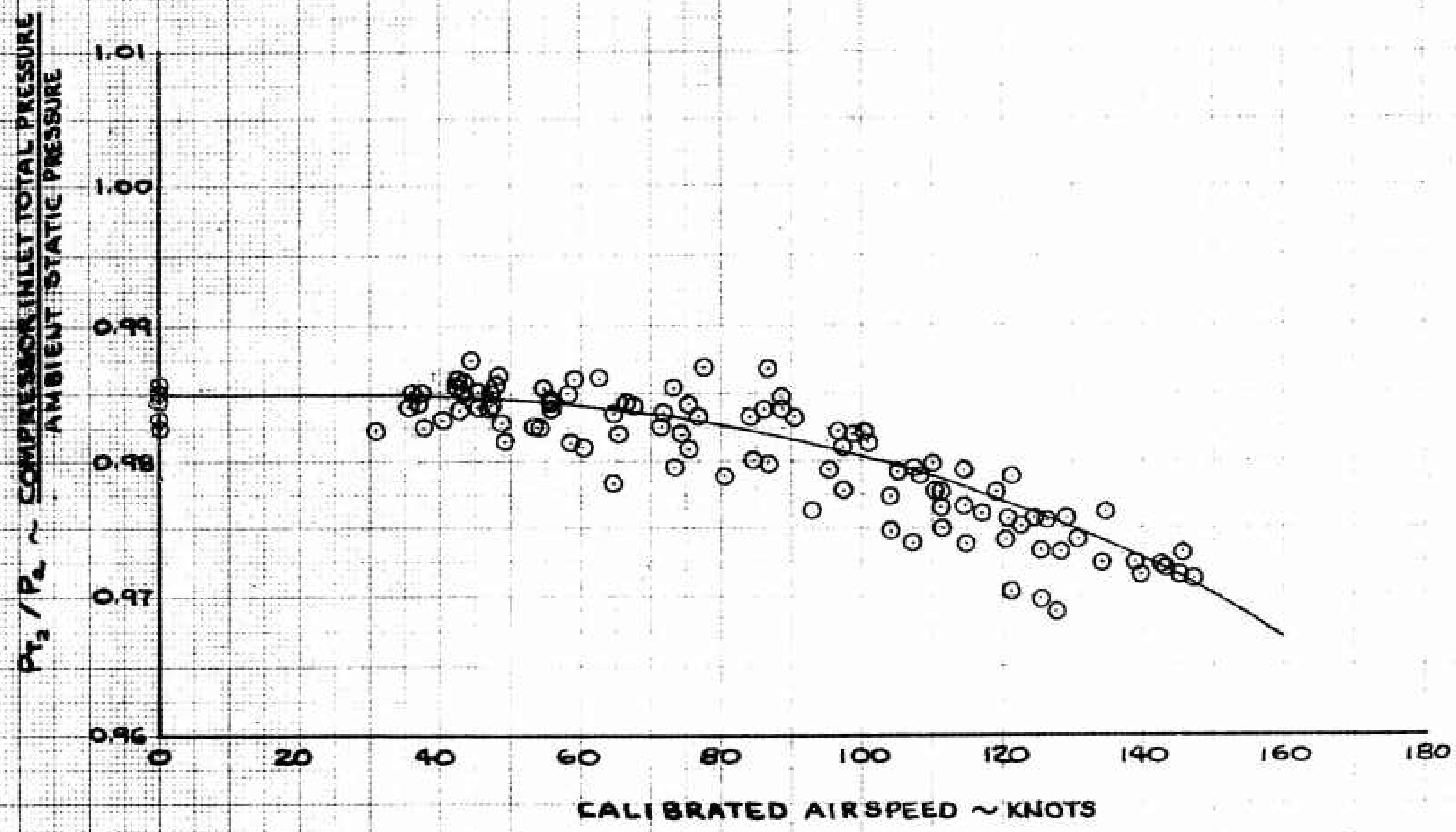
ENGINE CHARACTERISTICS

AH-1G USA S/N G15247

T53-L-13 S/N LE 14001

ENGINE PARTICLE SEPARATOR INSTALLED

NOTE : SHAFT HORSE POWER AVAILABLE BASED ON SOLID CURVE



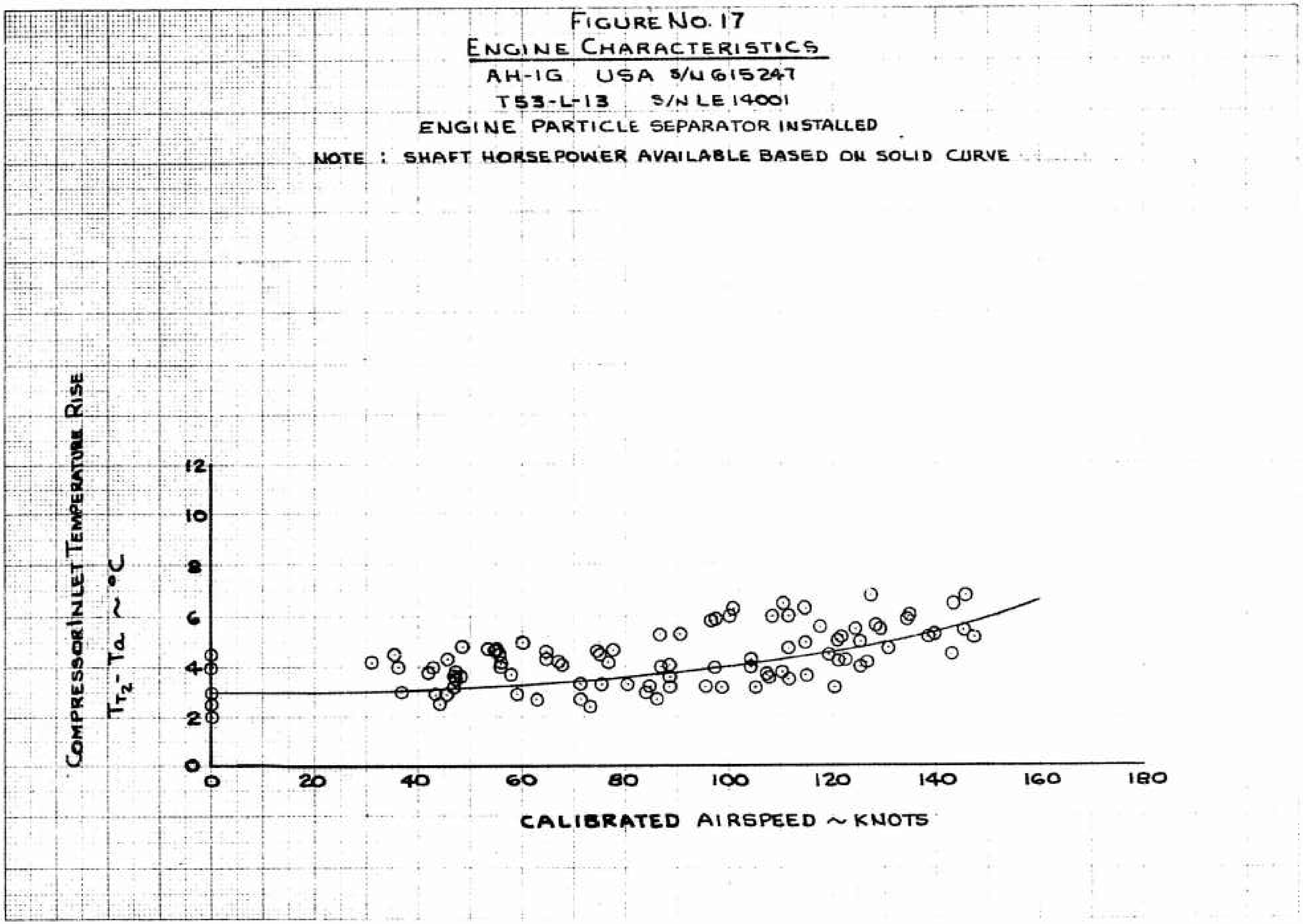


FIGURE NO. 17
ENGINE CHARACTERISTICS
AH-1G USA S/N 615247
T53-L-13 S/N LE 14001
ENGINE PARTICLE SEPARATOR INSTALLED
NOTE : SHAFT HORSEPOWER AVAILABLE BASED ON SOLID CURVE
COMPRESSOR INLET TEMPERATURE RISE
$T_{T_2} - T_a$ ~ °C
0
2
4
6
8
10
12
0
20
40
60
80
100
120
140
160
180
CALIBRATED AIRSPEED ~ KNOTS

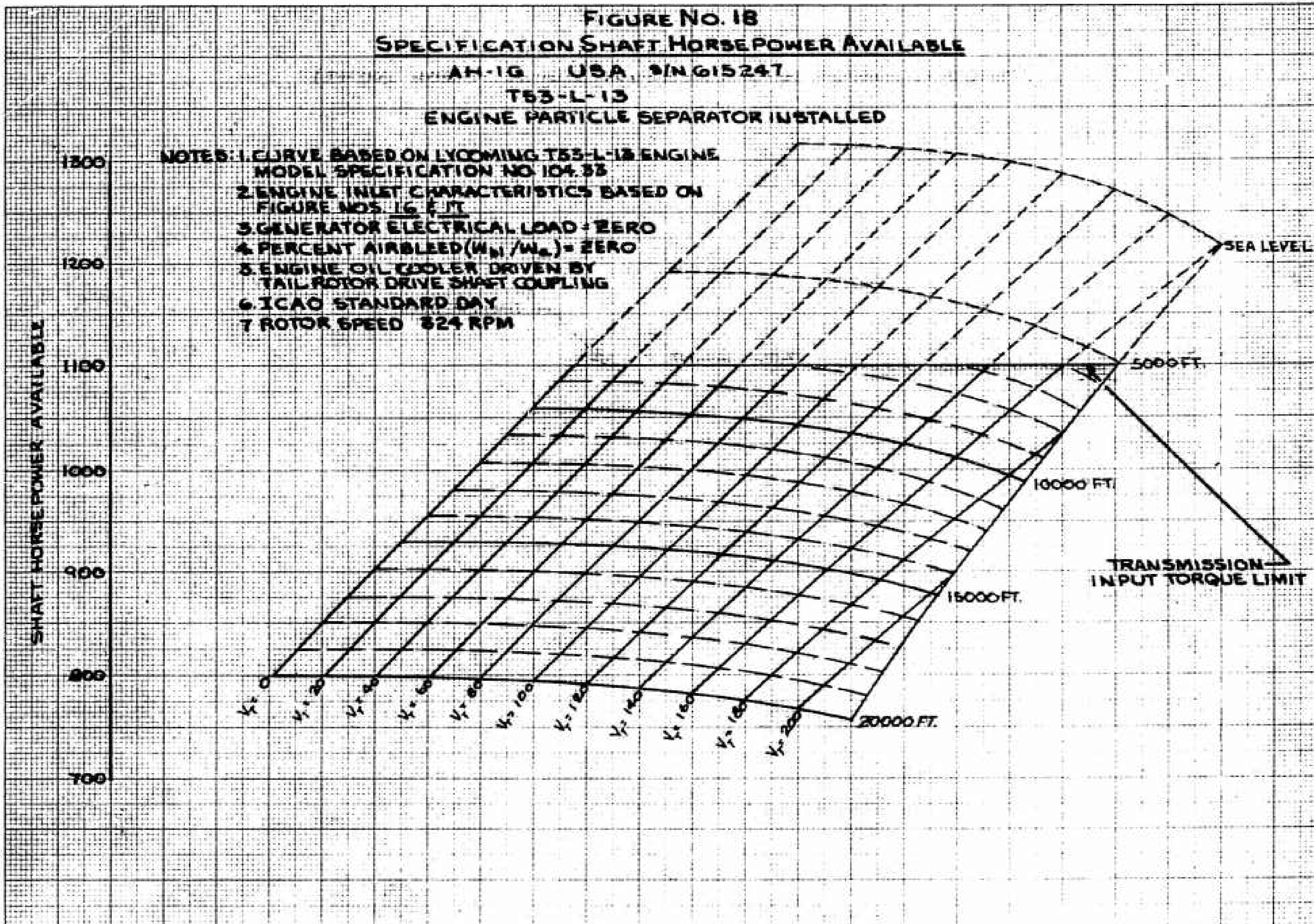


FIGURE NO. 18
SPECIFICATION SHAFT HORSEPOWER AVAILABLE
AH-1G USA S/N 615247
T53-L-13
ENGINE PARTICLE SEPARATOR INSTALLED
NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33
2. ENGINE INLET CHARACTERISTICS BASED ON FIGURE NOS. 16 & 17
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIRBLEED (W_{bl}/W_a) = ZERO
5. ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING
6. ICAO STANDARD DAY
7. ROTOR SPEED 324 RPM
SHAFT HORSEPOWER AVAILABLE
1300
1200
1100
1000
900
800
700
SEA LEVEL
5000 FT.
10000 FT.
15000 FT.
20000 FT.
TRANSMISSION INPUT TORQUE LIMIT
V_T = 0
V_T = 20
V_T = 40
V_T = 60
V_T = 80
V_T = 100
V_T = 120
V_T = 140
V_T = 160
V_T = 180
V_T = 200

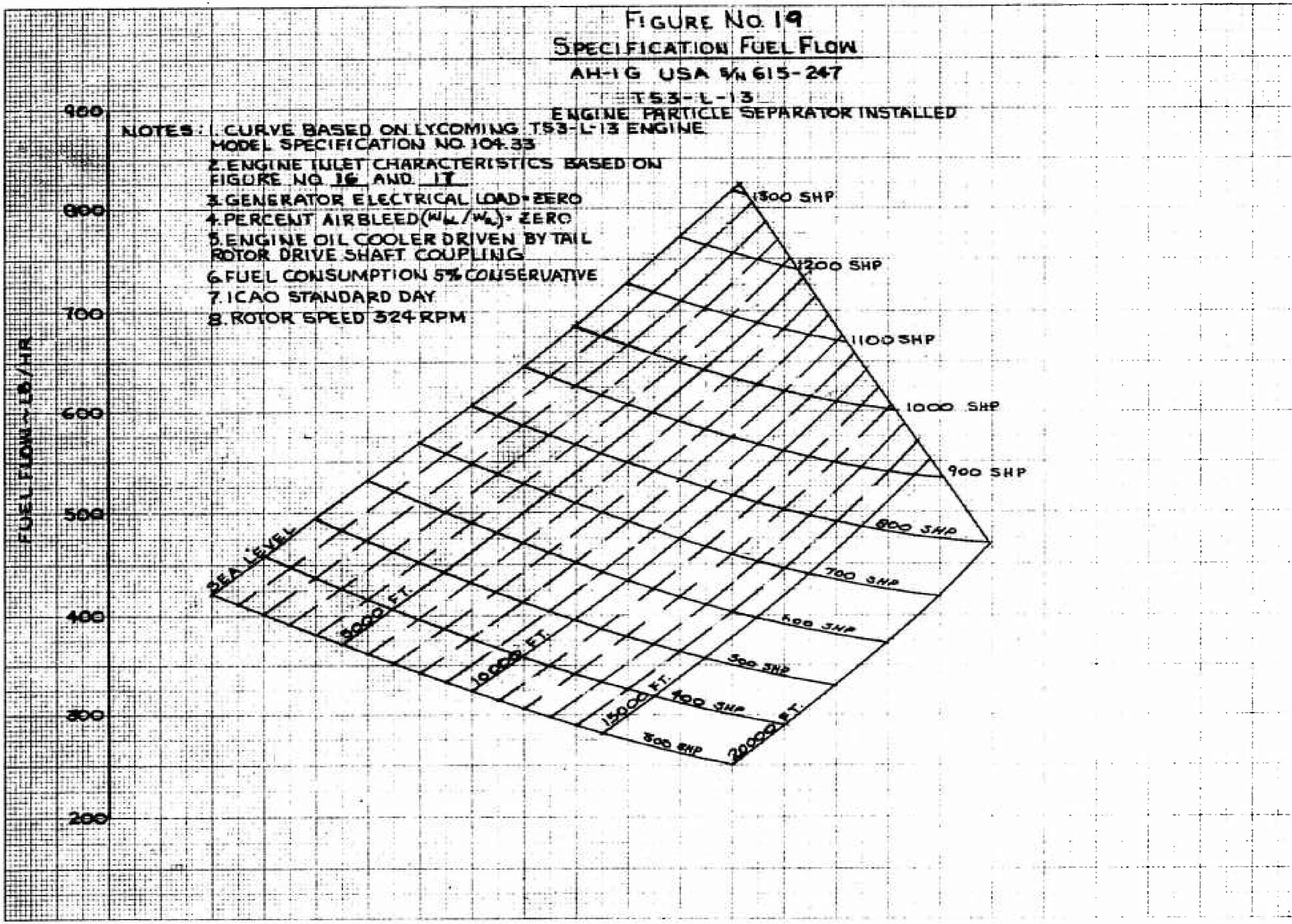


FIGURE No. 19
SPECIFICATION FUEL FLOW
AH-1G USA S/N 615-247
T53-L-13
ENGINE PARTICLE SEPARATOR INSTALLED
NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33
2. ENGINE INLET CHARACTERISTICS BASED ON FIGURE NO. 16 AND 17
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIRBLEED (W_L/W_A) = ZERO
5. ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING
6. FUEL CONSUMPTION 5% CONSERVATIVE
7. ICAO STANDARD DAY
8. ROTOR SPEED 324 RPM
FUEL FLOW ~ LB/HR
900
800
700
600
500
400
300
200
1300 SHP
1200 SHP
1100 SHP
1000 SHP
900 SHP
800 SHP
700 SHP
600 SHP
500 SHP
400 SHP
300 SHP
SEA LEVEL
5000 FT.
10000 FT.
15000 FT.
20000 FT.

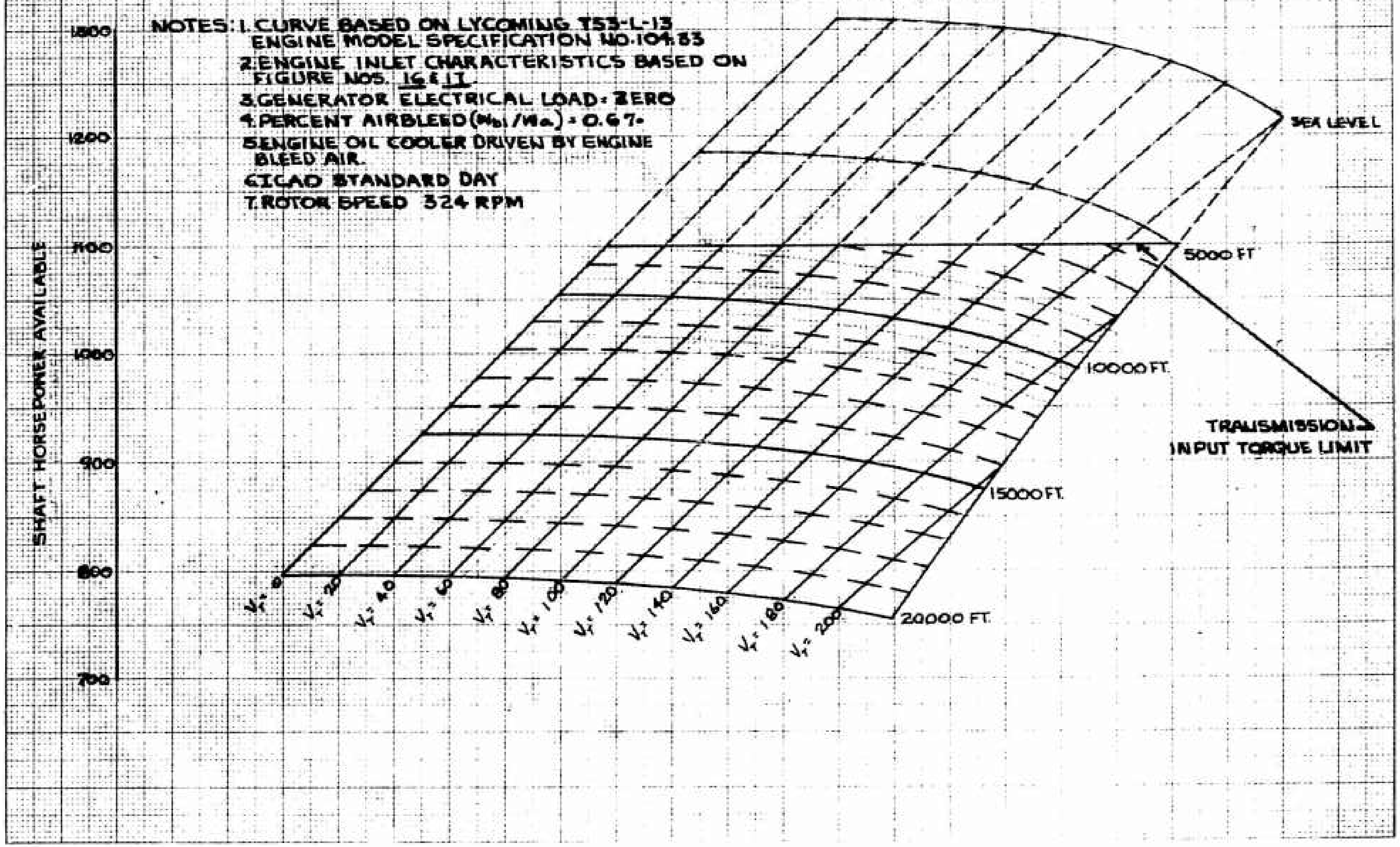
FIGURE NO. 20

SPECIFICATION SHAFT HORSEPOWER AVAILABLE

AH-1G USA S/N 615247

T53-L-13 S/N LE 14001

ENGINE PARTICLE SEPARATOR INSTALLED



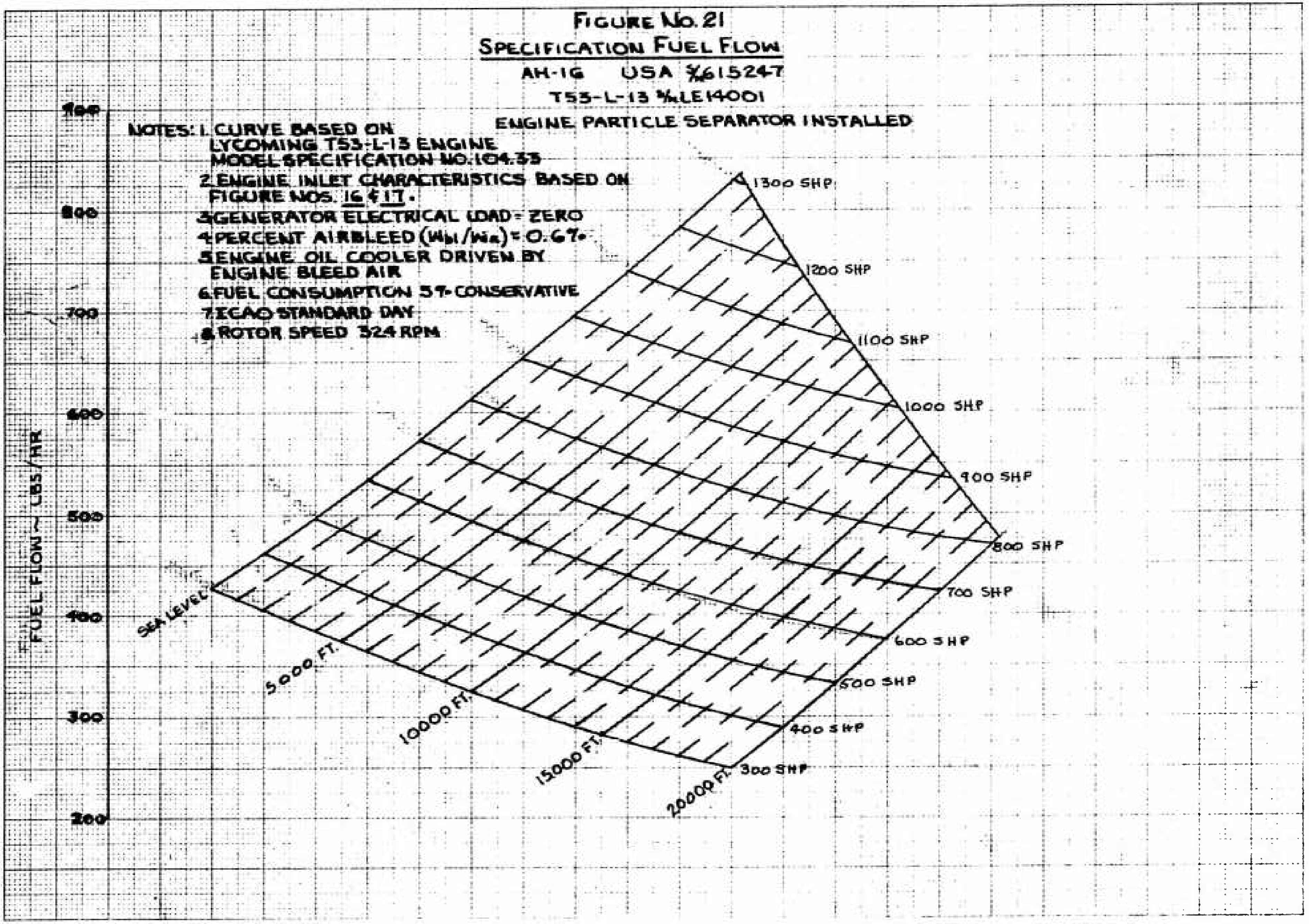


FIGURE No. 21
SPECIFICATION FUEL FLOW
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ENGINE PARTICLE SEPARATOR INSTALLED
NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO.104.33
2. ENGINE INLET CHARACTERISTICS BASED ON FIGURE NOS. 16 & 17.
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIRBLEED (Wbl/Wa) = 0.67
5. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR
6. FUEL CONSUMPTION 5% CONSERVATIVE
7. ICAO STANDARD DAY
8. ROTOR SPEED 324 RPM
FUEL FLOW ~ LBS/HR
900
800
700
600
500
400
300
200
1300 SHP
1200 SHP
1100 SHP
1000 SHP
900 SHP
800 SHP
700 SHP
600 SHP
500 SHP
400 SHP
300 SHP
SEA LEVEL
5000 FT.
10000 FT.
15000 FT.
20000 FT.

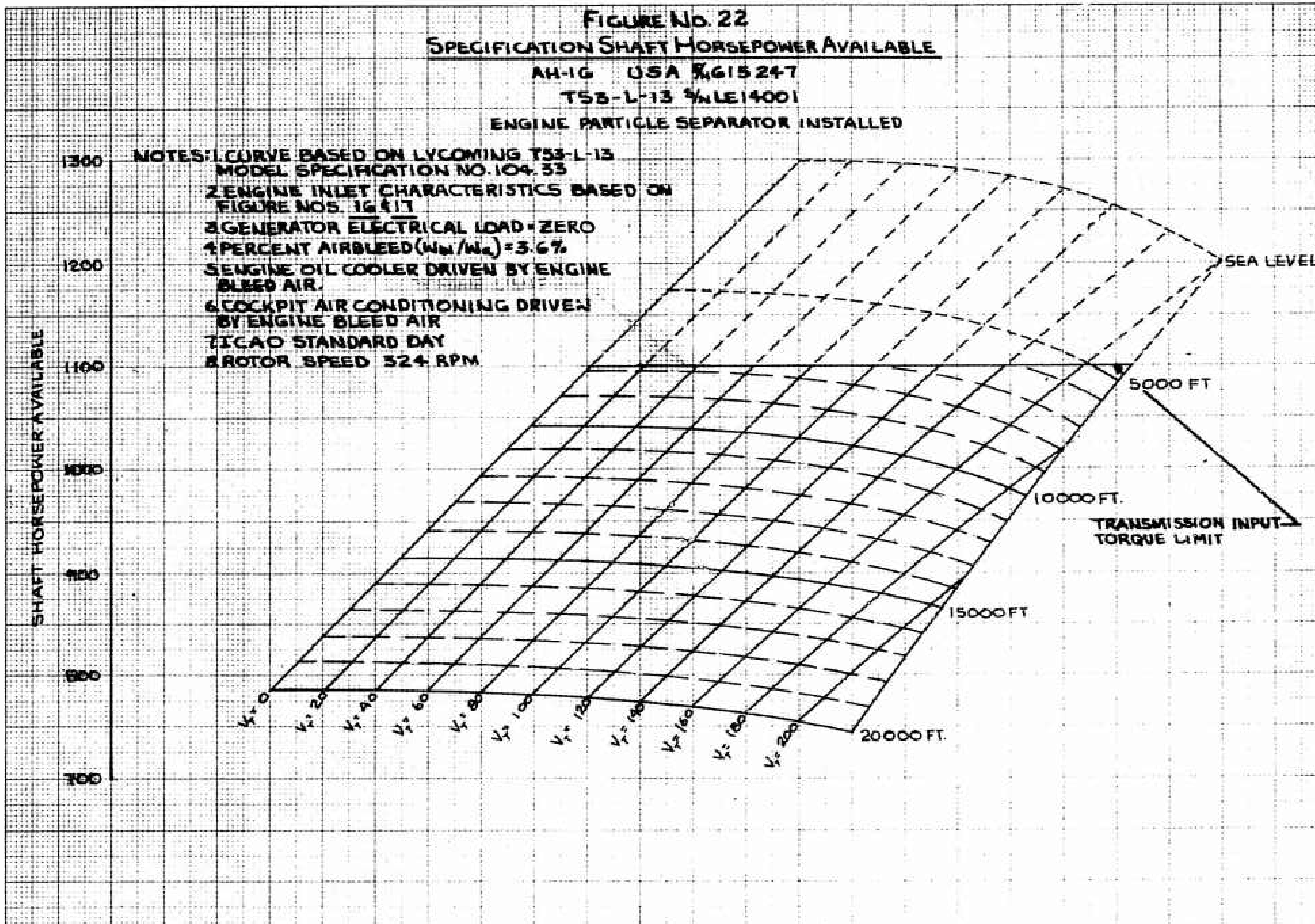


FIGURE NO. 22
SPECIFICATION SHAFT HORSEPOWER AVAILABLE
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ENGINE PARTICLE SEPARATOR INSTALLED
NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 MODEL SPECIFICATION NO. 104.33
2. ENGINE INLET CHARACTERISTICS BASED ON FIGURE NOS. 16 & 17
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIRBLEED (W_{bl}/W_a) = 3.6%
5. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR.
6. COCKPIT AIR CONDITIONING DRIVEN BY ENGINE BLEED AIR
7. ICAO STANDARD DAY
8. ROTOR SPEED 324 RPM
SHAFT HORSEPOWER AVAILABLE
1300
1200
1100
1000
900
800
700
SEA LEVEL
5000 FT
10000 FT.
15000 FT
20000 FT.
TRANSMISSION INPUT TORQUE LIMIT
V_T = 0
V_T = 20
V_T = 40
V_T = 60
V_T = 80
V_T = 100
V_T = 120
V_T = 140
V_T = 160
V_T = 180
V_T = 200

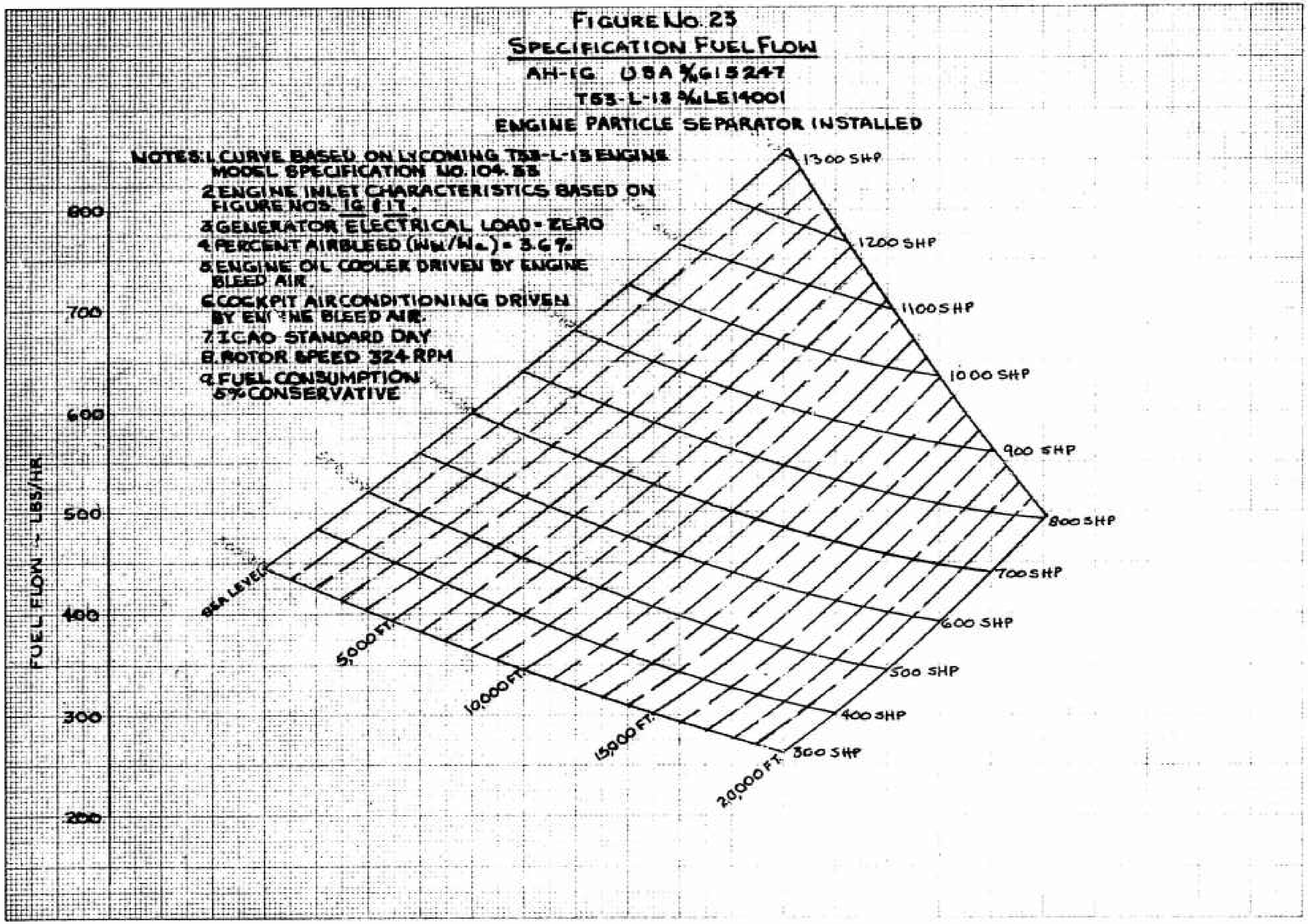


FIGURE No. 23
SPECIFICATION FUEL FLOW
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ENGINE PARTICLE SEPARATOR INSTALLED
NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.55
2. ENGINE INLET CHARACTERISTICS BASED ON FIGURE NOS. 16 & 17.
3. GENERATOR ELECTRICAL LOAD - ZERO
4. PERCENT AIRBLEED (Wb/Wa) = 3.6%
5. ENGINE OIL COOLER DRIVEN BY ENGINE BLEED AIR.
6. COCKPIT AIRCONDITIONING DRIVEN BY ENGINE BLEED AIR.
7. ICAO STANDARD DAY
8. ROTOR SPEED 324 RPM
9. FUEL CONSUMPTION 5% CONSERVATIVE
1300 SHP
1200 SHP
1100 SHP
1000 SHP
900 SHP
800 SHP
700 SHP
600 SHP
500 SHP
400 SHP
300 SHP
SEA LEVEL
5000 FT.
10,000 FT.
15,000 FT.
20,000 FT.
800
700
600
500
400
300
200
FUEL FLOW ~ LBS/HR

FIGURE No. 24
ENGINE CHARACTERISTICS
AH-1G USA S/N 615247

T53-L-13 S/N LE14001

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO.104.33

2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NOS. 16 & 17 FOR ZERO AIRSPEED

3. GENERATOR ELECTRICAL LOAD = ZERO.

4. PERCENT AIRBLEED (W_{bl}/W_a) = ZERO

5. ROTOR SPEED = 324 RPM

6. ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING

7. SOLID SYMBOLS DERIVED FROM ENGINE MANUFACTURE'S CALIBRATION CONDUCTED ON 22 AUGUST 1967

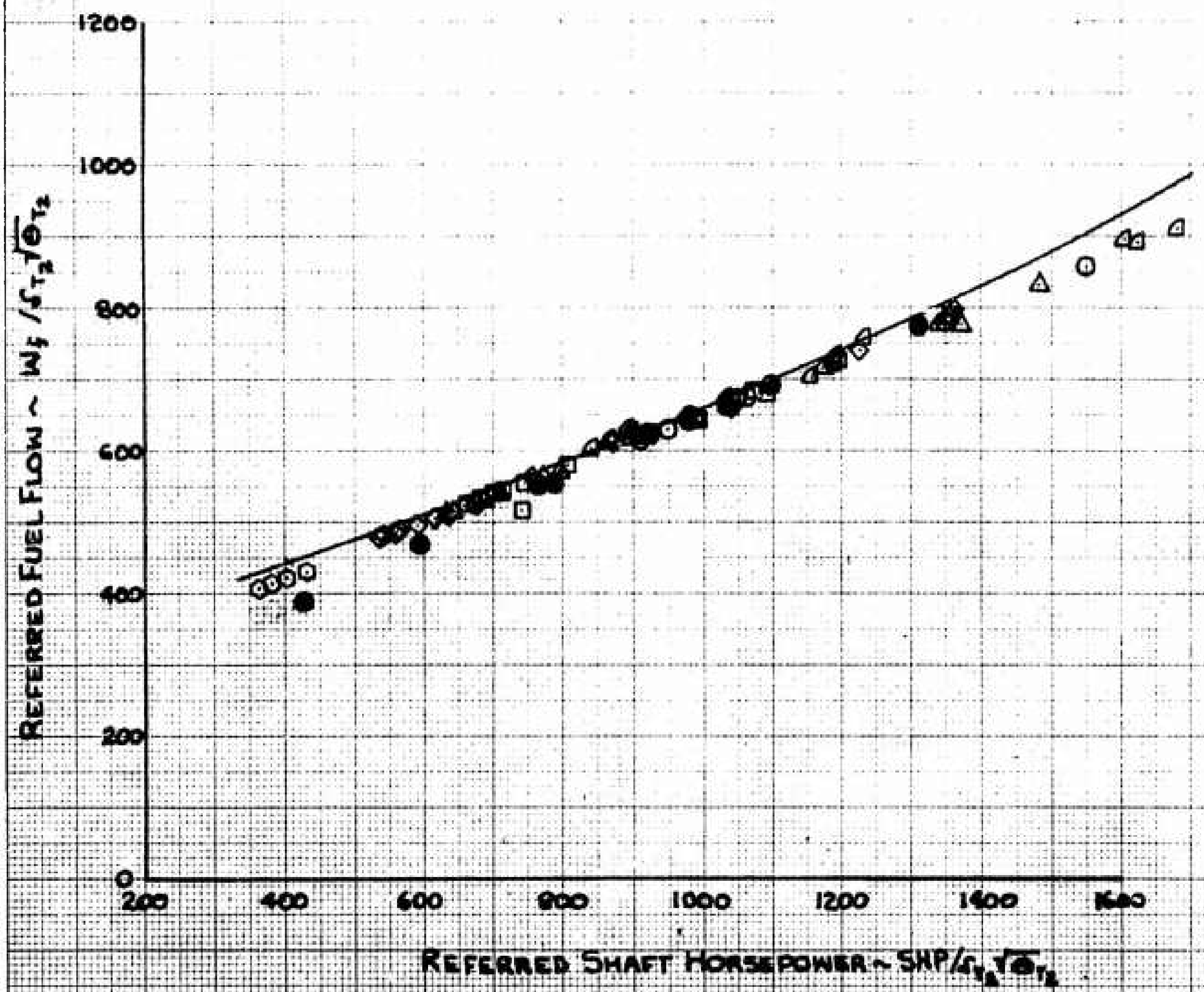


FIGURE NO. 25
ENGINE CHARACTERISTICS
AH-1G USA S/N 615247
T53-L-13 S/N LE14001
ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33.
2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NOS. 16 & 17 FOR ZERO AIRSPEED.
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIR BLEED (W_{bl}/W_a) = ZERO
5. ROTOR SPEED 324 RPM
6. ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING.
7. SOLID SYMBOLS DERIVED FROM ENGINE MANUFACTURE'S CALIBRATION CONDUCTED ON 22 AUGUST 1967

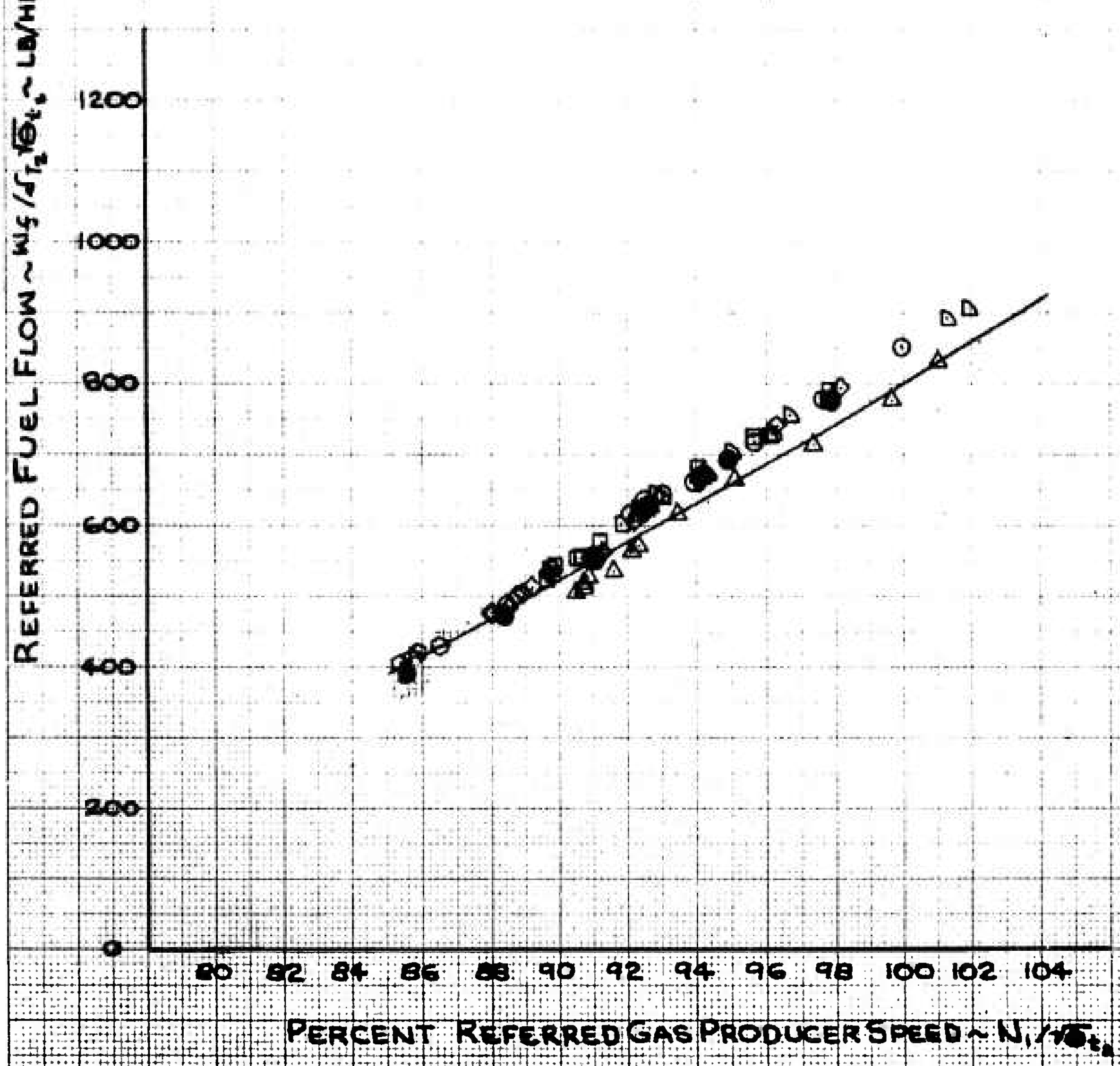


FIGURE No. 26

ENGINE CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES:
1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33
2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NOS. 16 & 17 FOR ZERO AIRSPEED
3. GENERATOR ELECTRICAL LOAD = ZERO
4. PERCENT AIRBLEED (W_b/W_a) = ZERO
5. ROTOR SPEED = 324 RPM
6. ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING
7. SOLID SYMBOLS DERIVED FROM ENGINE MANUFACTURE'S CALIBRATION CONDUCTED ON 22 AUGUST 1967

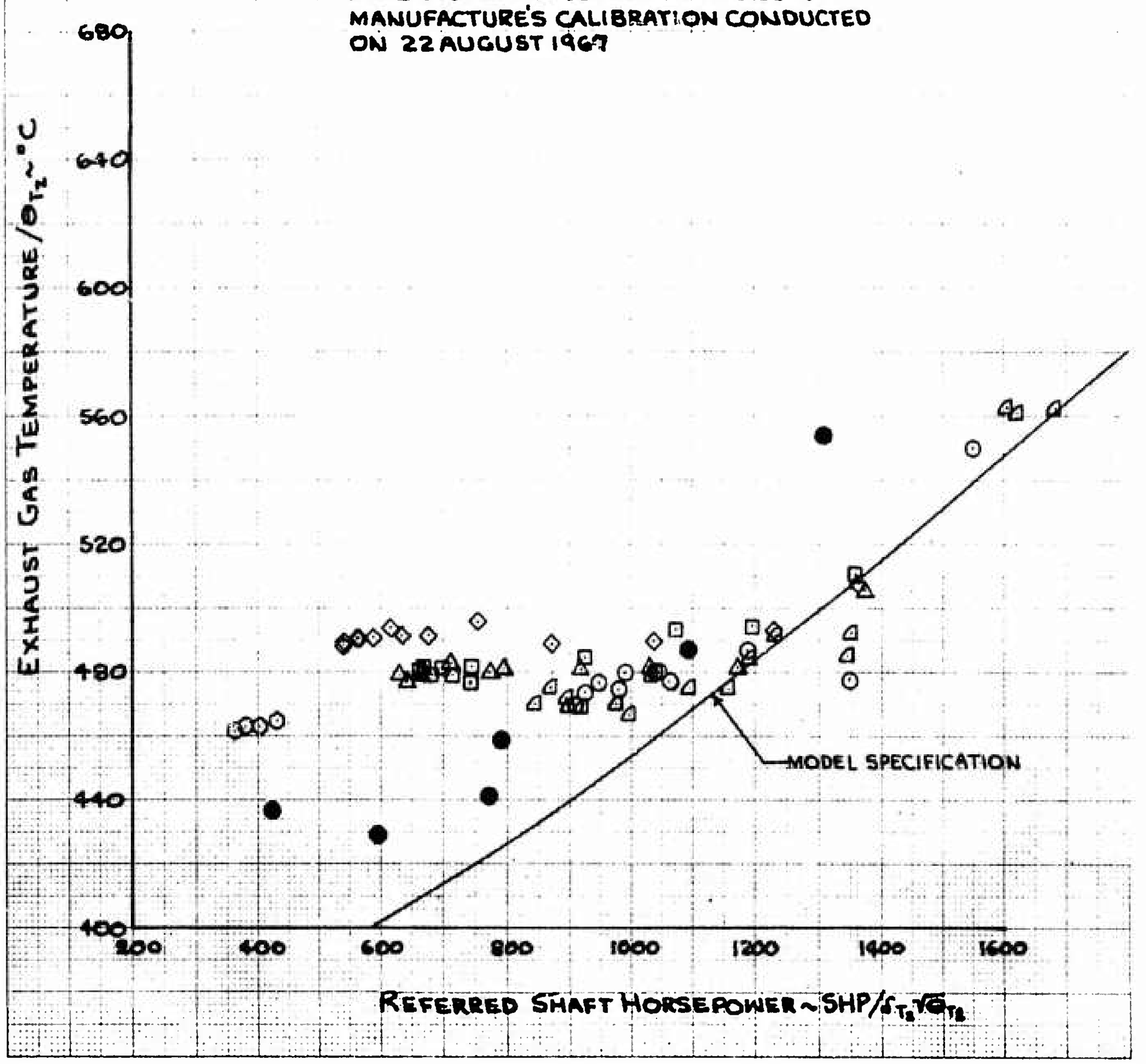


FIGURE NO 27

ENGINE CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33

2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NOS. 16 & 17 FOR ZERO AIRSPEED

3. GENERATOR ELECTRICAL LOAD = ZERO

4. PERCENT AIRBLEED (W_{bl}/W_a) = ZERO

5. ROTOR SPEED = 324 RPM

6. ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING

7. SOLID SYMBOLS DERIVED FROM ENGINE MANUFACTURE'S CALIBRATION CONDUCTED ON 22 AUGUST 1967

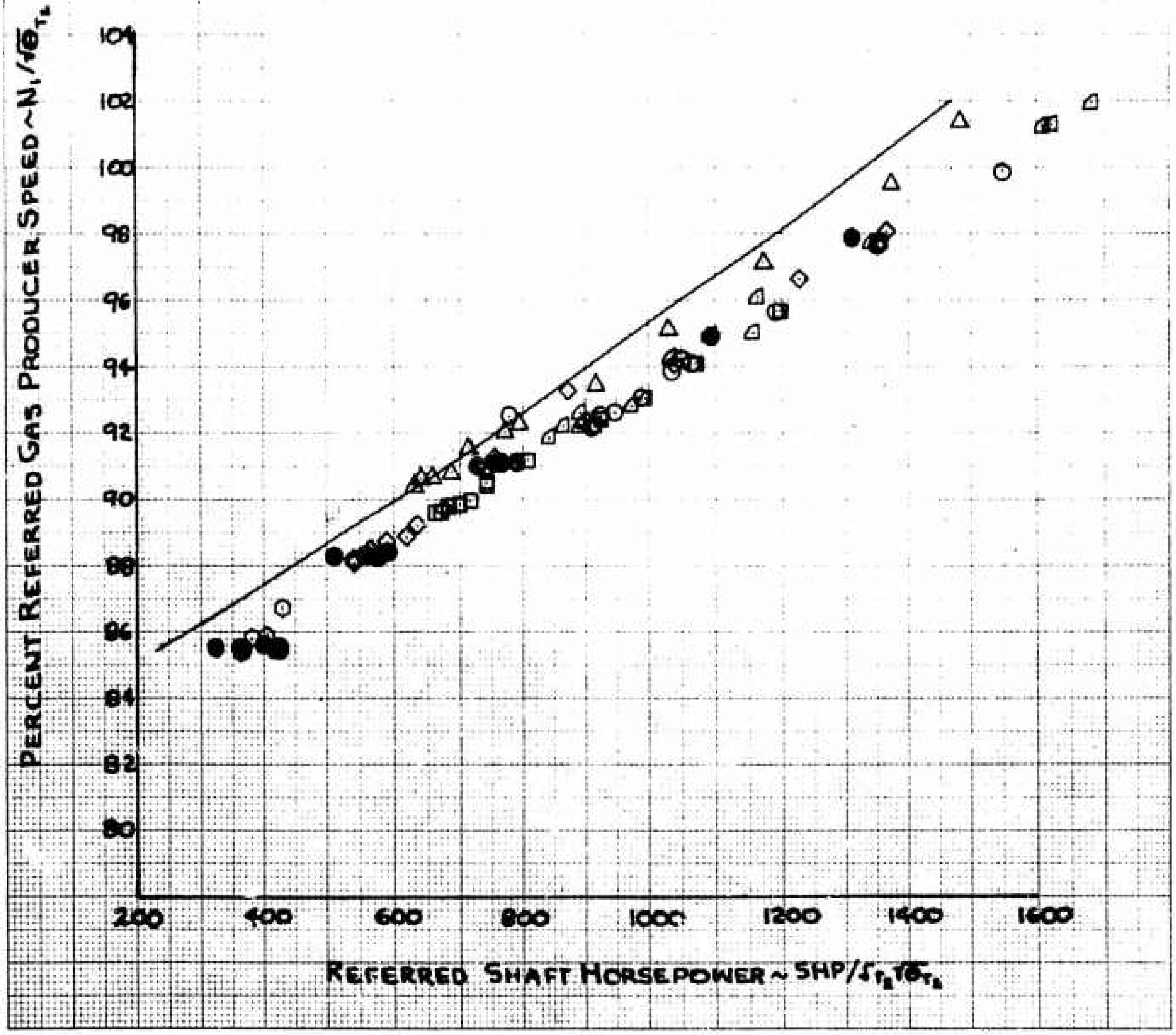


FIGURE NO. 28

ENGINE CHARACTERISTICS

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

ENGINE PARTICLE SEPARATOR INSTALLED

NOTES: 1. CURVE BASED ON LYCOMING T53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33

2. CURVE BASED ON ENGINE INLET CHARACTERISTICS PRESENTED IN FIGURE NOS. 16 & 17 FOR ZERO AIRSPEED

3. GENERATOR ELECTRICAL LOAD = ZERO

4. PERCENT AIRBLEED (W_{bl}/W_a) = ZERO

5. ROTOR SPEED = 324 RPM

6. ENGINE OIL COOLER DRIVEN BY TAIL ROTOR DRIVE SHAFT COUPLING

7. SOLID SYMBOLS DERIVED FROM ENGINE MANUFACTURE'S CALIBRATION CONDUCTED ON 22 AUGUST 1967

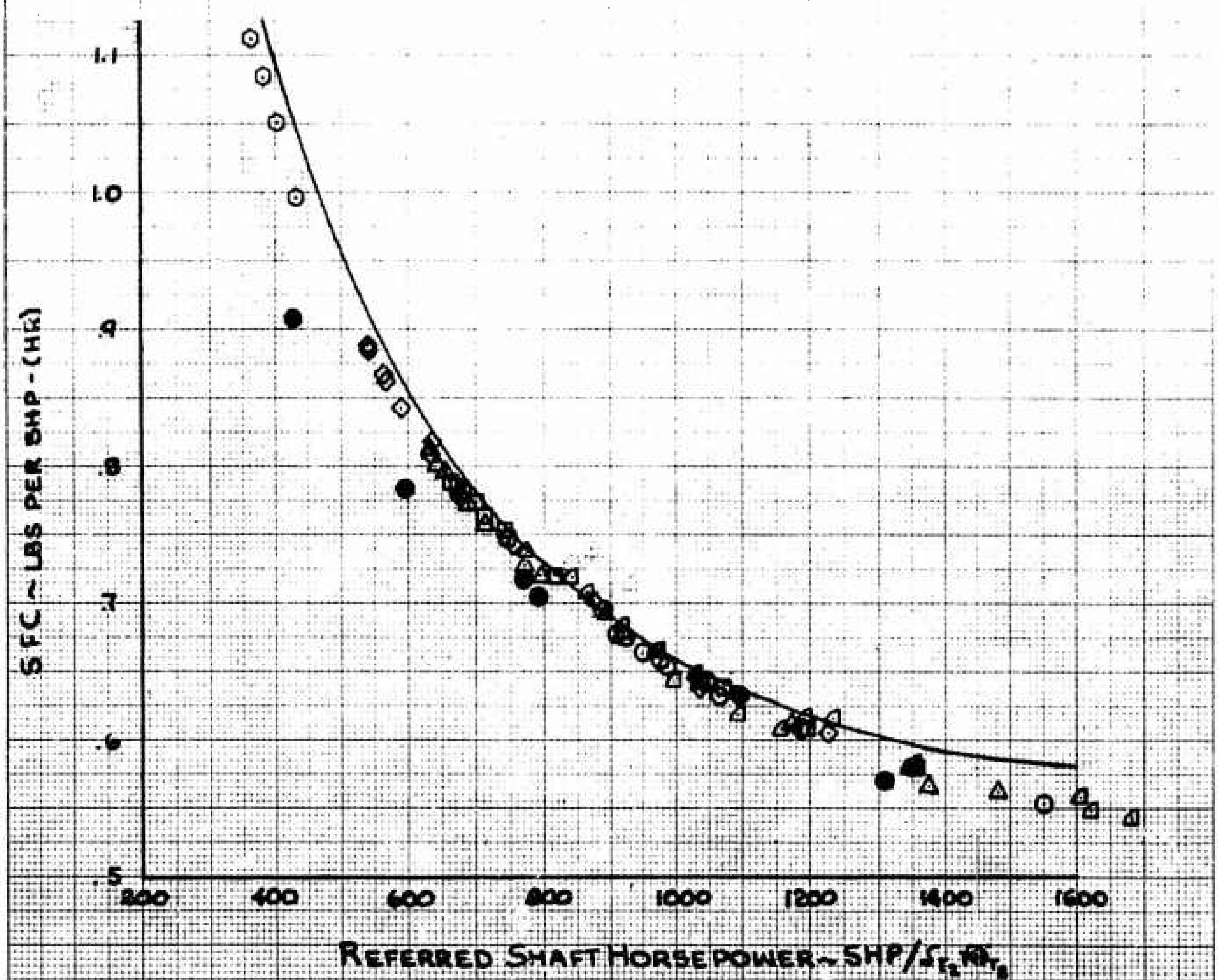


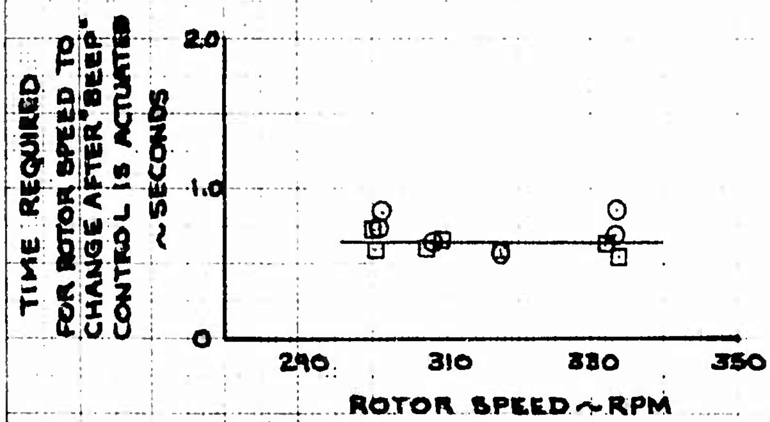
Figure No. 29

Engine "Beep" Control Characteristics

AH-1G USA S/N 615247

T53-L-13 S/N LE14001

Symbol	Airspeed ~ KCAS	Density Alt. ~ FT.	Flight Condition
○	105	5000	Level Flight
□	Zero	1500	Ground Run (Collective Full Down)



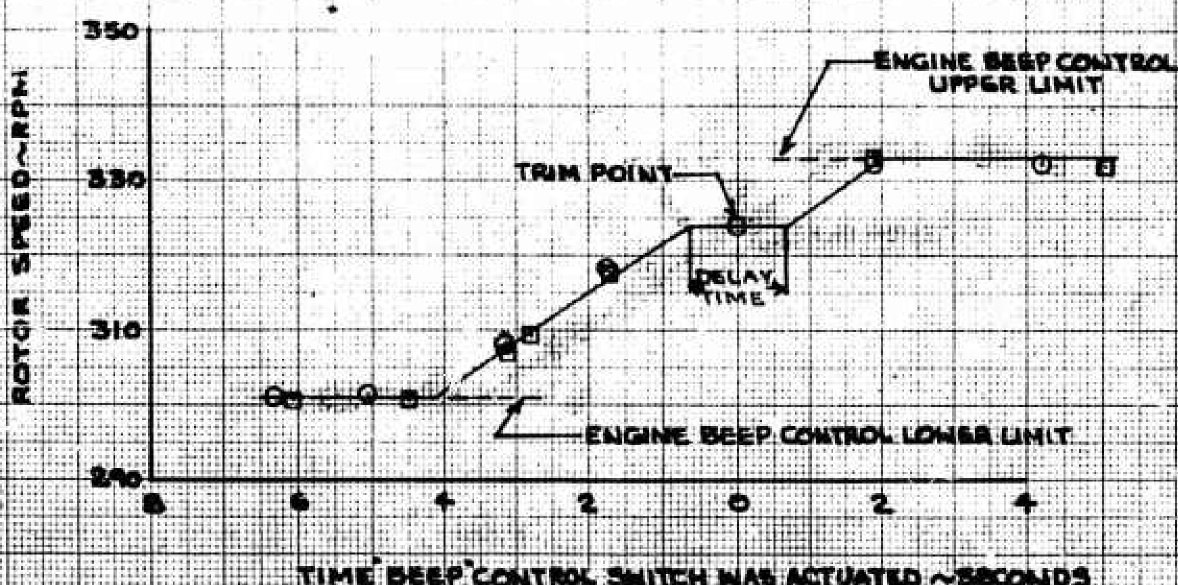


FIGURE No. 30

TAIL ROTOR PITCH REQUIRED WITH WIND AZIMUTH 160°

AH-1G USA S/N 615247

OUTBOARD ALTERNATE CONFIGURATION (FAIRED)

NOTES:
1. STANDARD TAIL ROTOR
2. FULL LEFT PEDAL = 19° TAIL ROTOR PITCH WITH SCAS NULL
3. GROUND SPEED DETERMINED WITH CALIBRATED PACE CAR
4. TRUE AIRSPEED DETERMINED BY VECTOR ADDITION OF GROUND SPEED AND WIND
5. SOLID SYMBOL DENOTES DATA DERIVED FROM USAAVNTA REPORT NO 66-06
6. TOTAL DIRECTIONAL CONTROL DISPLACEMENT = 7.07 IN. FROM FULL LEFT

△ MEAN TAIL ROTOR PITCH

○ MAXIMUM TAIL ROTOR PITCH

DOORS ON

DENSITY ALTITUDE = 1040 FT.

GROSS WEIGHT = 8075 LBS.

CENTER OF GRAVITY = 195.6 IN. (MID)

ROTOR SPEED = 324 RPM

DOORS OFF

DATA NOT AVAILABLE FOR THIS CONFIGURATION

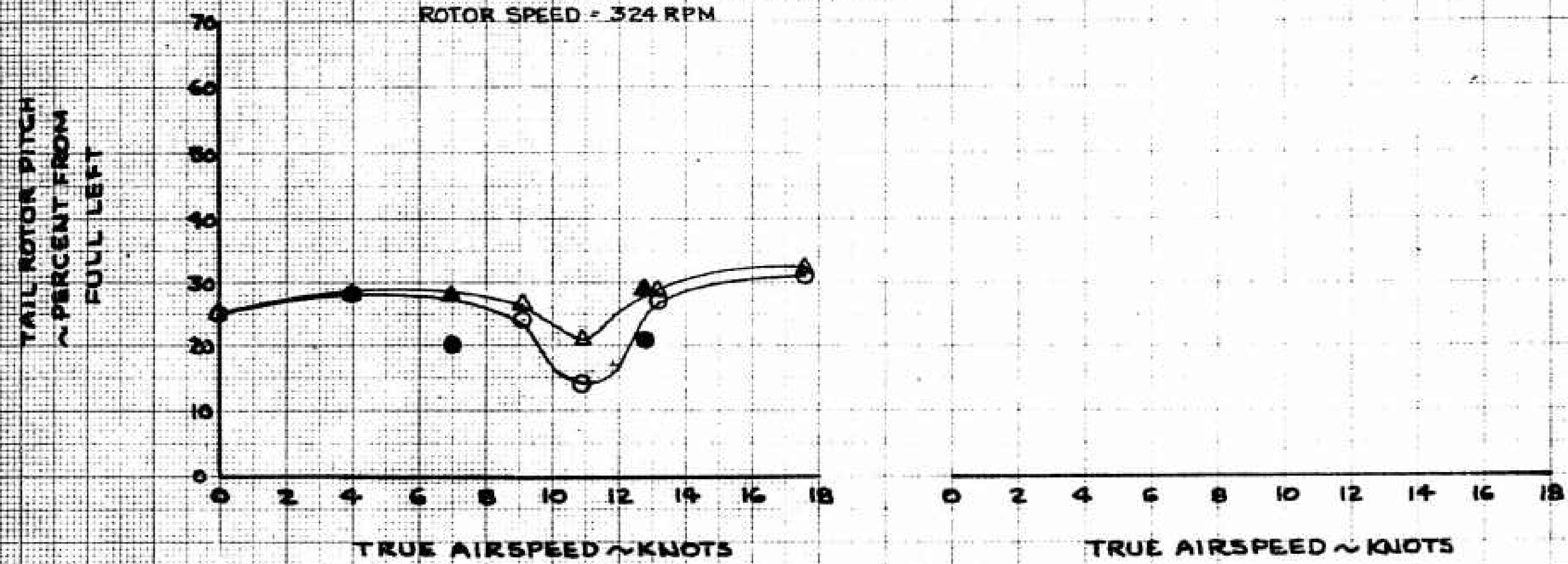


FIGURE No. 31

TAIL ROTOR PITCH REQUIRED WITH WIND AZIMUTH 200°

AH-1G USA S/N 615247

OUTBOARD ALTERNATE CONFIGURATION (FAIRED)

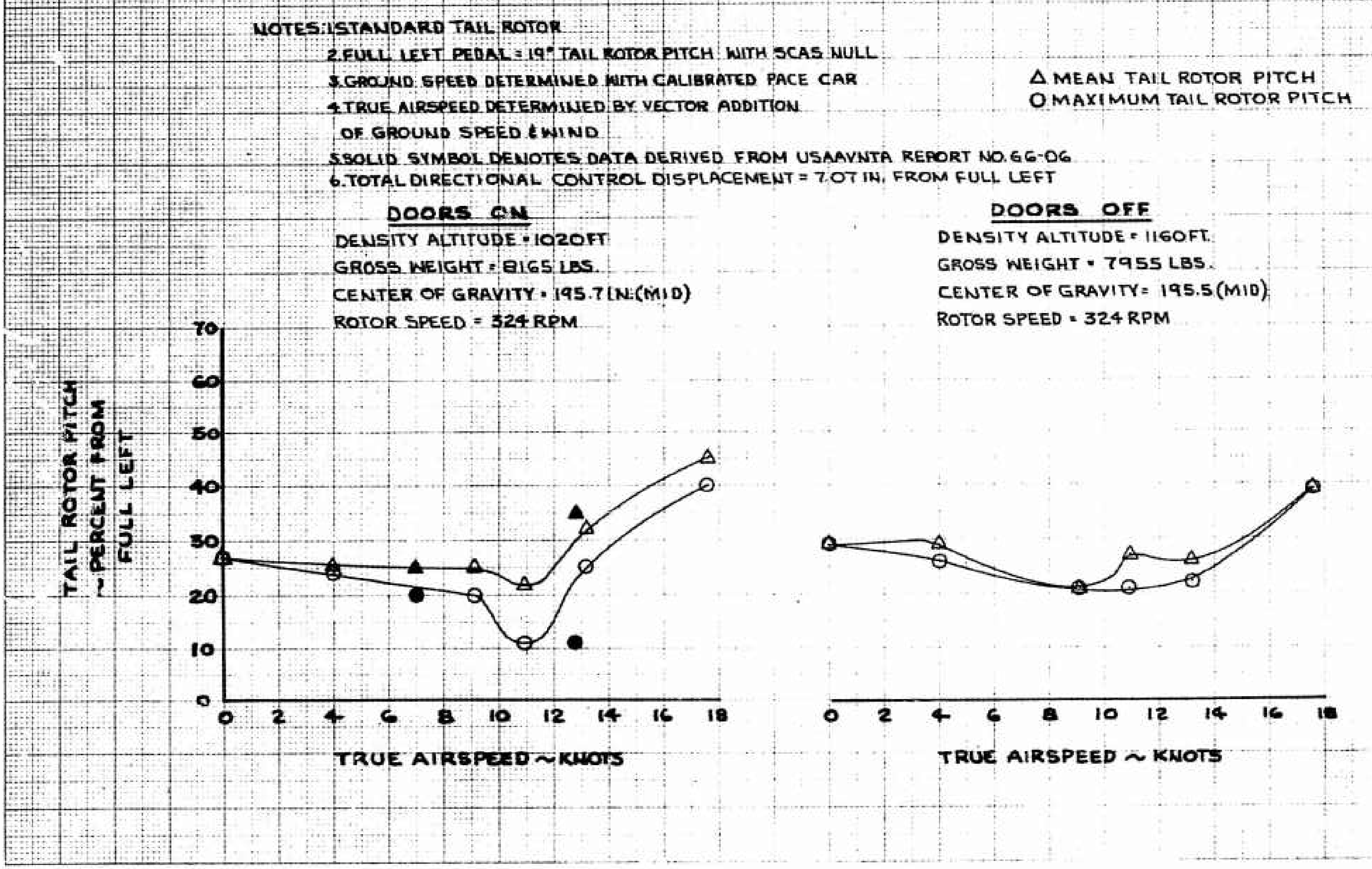


FIGURE No. 32

TAIL ROTOR PITCH REQUIRED WITH WIND AZIMUTH 240°

AH-1G USA S/N 615247

OUTBOARD ALTERNATE CONFIGURATION (FAIRED)

NOTES: 1. STANDARD TAIL ROTOR
2. FULL LEFT PEDAL = 19° TAIL ROTOR WITH SCAS NULL
3. GROUND SPEED DETERMINED WITH CALIBRATED PACE CAR
4. TRUE AIRSPEED DETERMINED BY VECTOR ADDITION OF GROUND SPEED & WIND
5. SOLID SYMBOL DENOTES DATA DERIVED FROM USAAVNTA REPORT NO. 66-06 (REF. 8, APP. I)
6. TOTAL DIRECTIONAL CONTROL DISPLACEMENT = 7.07 IN. FROM FULL LEFT

△ MEAN TAIL ROTOR PITCH
○ MAXIMUM TAIL ROTOR PITCH

DOORS ON
DENSITY ALTITUDE = 900 FT.
GROSS WEIGHT = 8260 LBS.
CENTER OF GRAVITY = 195.8 IN. (MID)
ROTOR SPEED = 324 RPM

DOORS OFF
DENSITY ALTITUDE = 1220 FT.
GROSS WEIGHT = 7860 LBS.
CENTER OF GRAVITY = 195.4 (MID)
ROTOR SPEED = 324 RPM

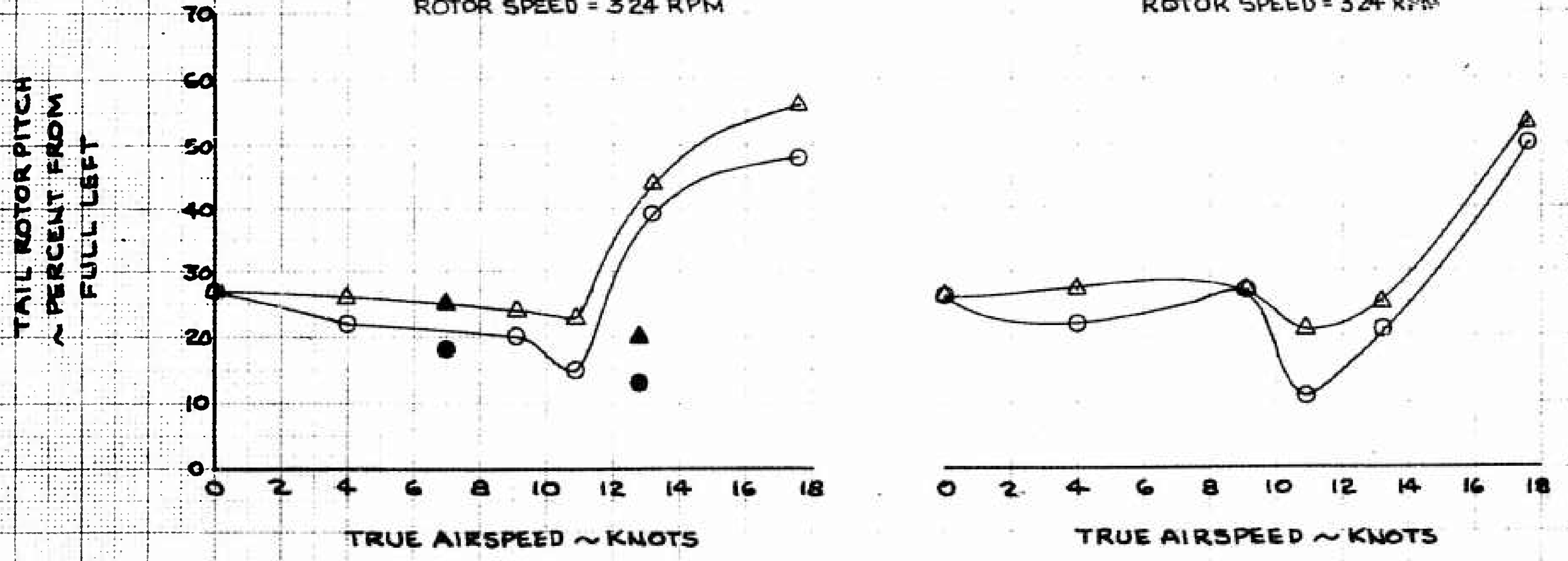
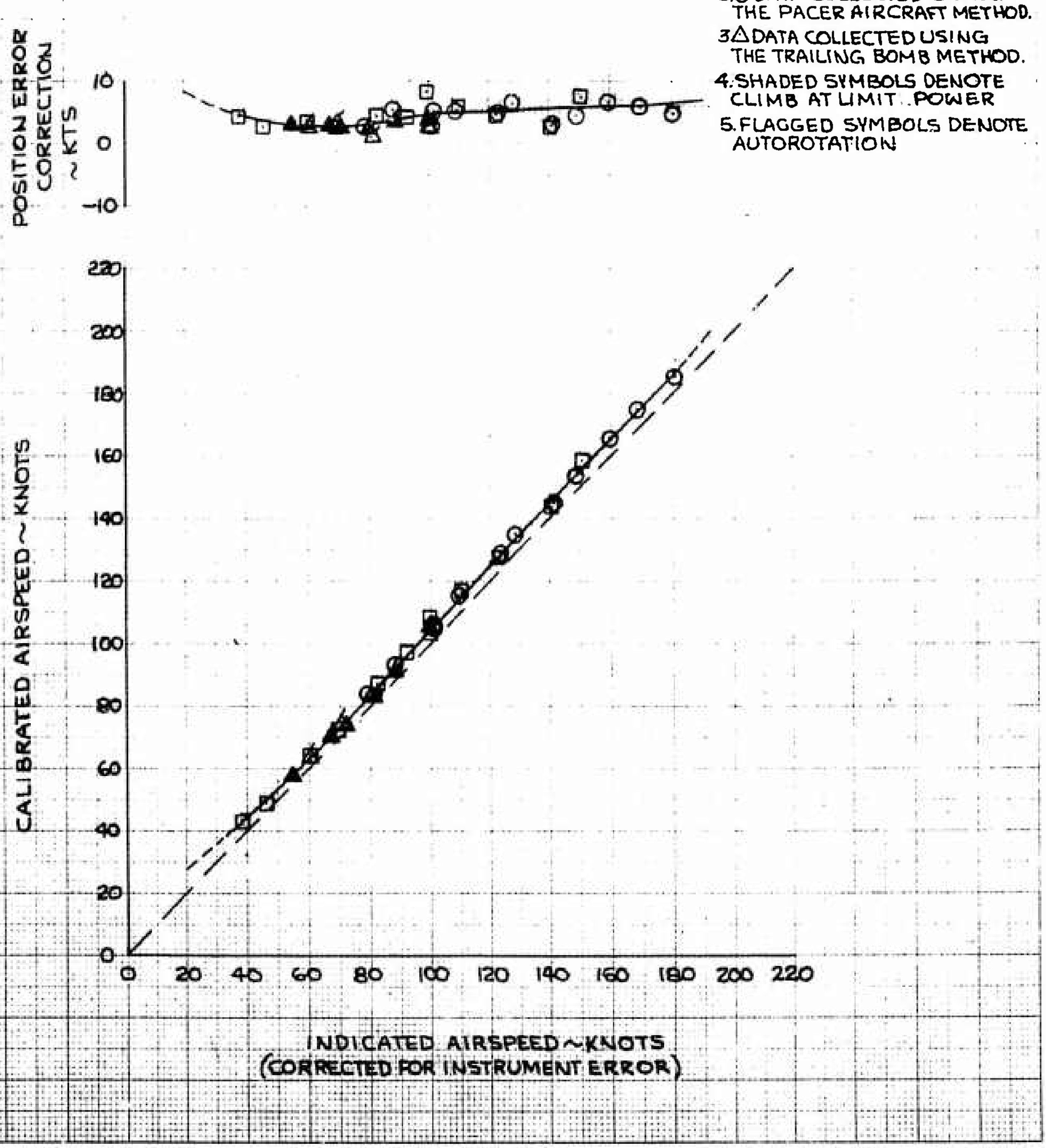


FIGURE No. 33
AIRSPEED CALIBRATION
AH-1G USA S/N 615247
BOOM SYSTEM

SYM.	GROSS WEIGHT ~LBS.	C.G. STATION ~IN.	DENSITY ALTITUDE ~FT.	ROTOR SPEED ~RPM	CONFIGURATION
⊡	7265	193.5	1020 FT.	324	CLEAN
⊙	7176	193.3	5300 FT.	324	CLEAN
△	7200	193.3	5000 FT	324	CLEAN

NOTES: 1. ⊡ DATA COLLECTED USING THE GROUND SPEED METHOD.
2. ⊙ DATA COLLECTED USING THE PACER AIRCRAFT METHOD.
3. △ DATA COLLECTED USING THE TRAILING BOMB METHOD.
4. SHADED SYMBOLS DENOTE CLIMB AT LIMIT POWER
5. FLAGGED SYMBOLS DENOTE AUTOROTATION



APPENDIX III. TEST INSTRUMENTATION

Flight test instrumentation was installed in the test helicopter prior to the start of this evaluation. This instrumentation provided data from four sources; pilot's panel, copilot/gunner's panel, photopanel and a 24-channel oscillograph. All instrumentation was calibrated. The flight test instrumentation was installed and maintained by the Instrumentation Branch, Logistics Division, USAASTA.

PILOT'S PANEL

Standard system airspeed
Boom system airspeed
Boom system altitude
Rate of climb
Gas producer speed
Torque pressure (standard system)
Exhaust gas temperature
Longitudinal control position
Lateral control position
Pedal control position
Collective control position
Center of gravity normal acceleration
Angle of sideslip

ENGINEER'S PANEL

Boom system airspeed
Boom system altitude
Outside air temperature
Rotor speed
Gas producer speed
Fuel used total
Torque pressure (high)
Torque pressure (low)
Exhaust gas temperature
Oscillograph correlation counter
Photopanel correlation counter
Fuel temperature

PHOTOPANEL

Boom system airspeed
Standard system airspeed
Boom system airspeed
Rotor speed
Gas producer speed
Fuel used total
Torque pressure (high)
Torque pressure (low)
Exhaust gas temperature
Compressor inlet temperature
Compressor inlet total pressure
Inlet guide vane position
Bleed band position (light)
Fuel pressure at nozzle
Time (10-second stopwatch)
Oscillograph correlation counter
Photopanel correlation counter
Engineer's event
Pilot's event

OSCILLOGRAPH

Longitudinal control position
Lateral control position
Directional control position
Collective control position
Pitch attitude
Roll attitude
Yaw attitude
Pitch rate
Roll rate
Yaw rate
CG normal acceleration
Angle of sideslip
Angle of attack
Engineer's event
Pilot's event
Photopanel correlation blip

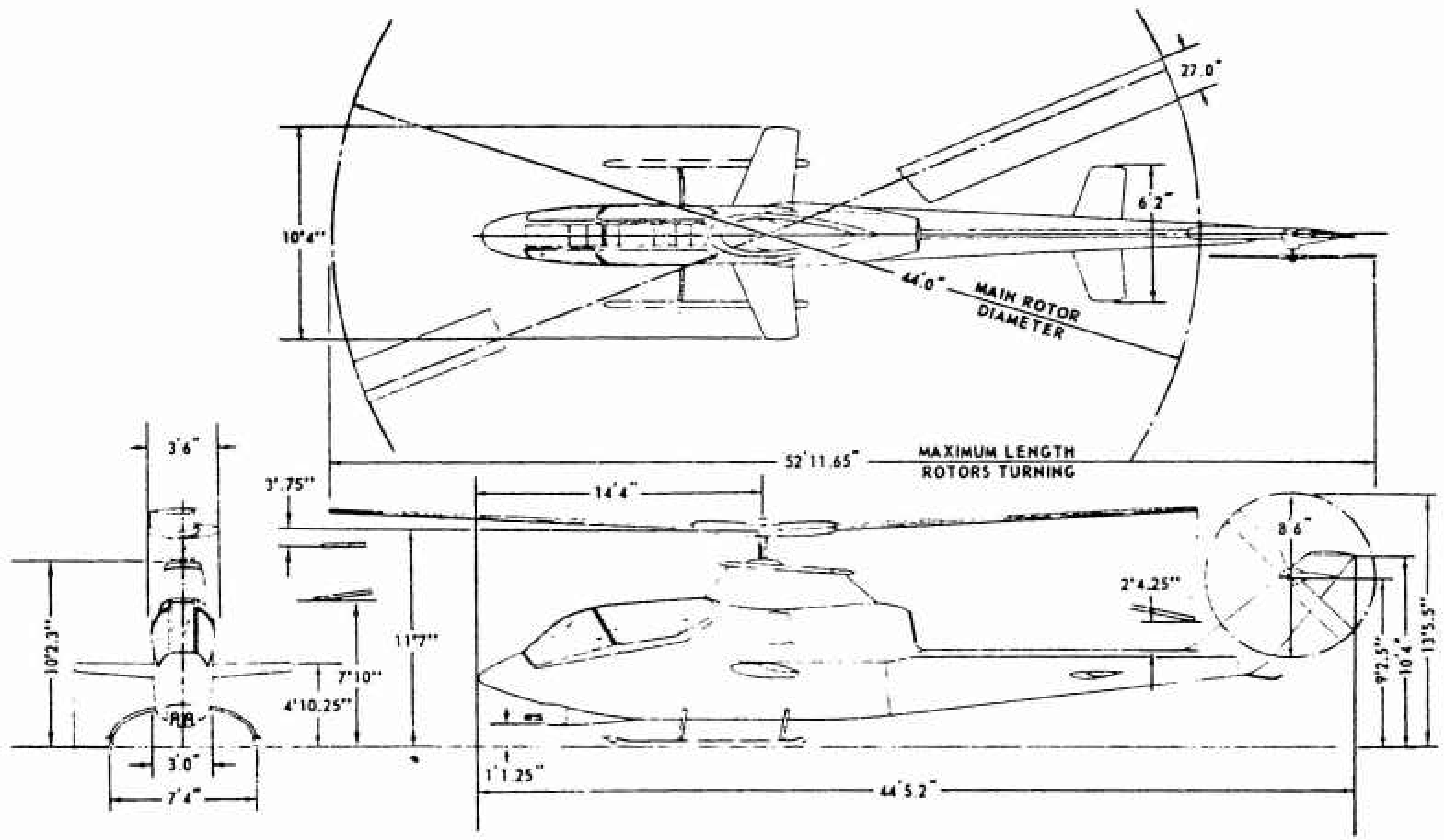
APPENDIX IV. BASIC AIRCRAFT DATA & OPERATING LIMITS

AIRFRAME DATA

Overall length (rotor turning)	637.2 inches
Overall width (rotor trailing)	124.0 inches
Center line of main rotor to center line of tail rotor	320.7 inches
Center line of main rotor to elevator hinge line	198.6 inches
Elevator area (total)	15.2 square feet
Elevator area (both panels)	10.9 square feet
Elevator airfoil section	Inverted Clark Y
Vertical stabilizer area	18.5 square feet
Vertical stabilizer airfoil section	Special chamber
Vertical stabilizer aero. center	Fuselage Station (FS) 499.0
Wing area:	
Total	27.8 square feet
Outboard of B.L. 18.0 (both sides)	18.5 square feet
Wing span	10.33 feet
Wing airfoil section:	
Root	NACA 0030
Tip	NACA 0024
Wing angle of incidence	14 degrees

MAIN ROTOR DATA

Number of blades	2
Diameter	44 feet
Disc area	1520.4 square feet
Blade chord	27 inches
Rotor solidity	0.065
Blade area (both blades)	99 square feet
Blade airfoil	9.33 percent symm. Section special
Blade twist	-0.455 deg/ft
Hub precone angle	2.75 degrees



Three-view drawing - AH-1G

ANTITORQUE ROTOR DATA

Number of blades	2
Diameter	8.5 feet
Disc area	56.74 square feet
Blade chord	8.41 inches
Rotor solidity	0.105
Blade airfoil	NACA 0010 modified
Blade twist	Zero degrees

TRANSMISSION DRIVE SYSTEM RATIOS

Engine to main rotor	20.383:1.0
Engine to antitorque rotor	3.990:1.0
Engine to antitorque drive system	1.535:1.0

LIMIT AIRSPEED (V_L)

Any configuration with XM159 rocket pods: 180 KCAS below a 3000-foot density altitude; decrease 8 KCAS per 1C feet above 3000 feet

All other configurations: 190 KCAS below a 4000-foot density altitude; decrease 8 KCAS per 1000 feet above 4000 feet

GROSS WEIGHT/CENTER OF GRAVITY ENVELOPE

Forward center of gravity limit: Below 7000 pounds, FS 190.0; linear increase to FS 192.1 at 9500 pounds

Aft center of gravity limit: Below 7880 pounds FS 201.4; linear decrease to FS 200 at 9500 pounds

SIDESLIP LIMITS

Five degrees at V_L with linear increase at 20 degrees at 60 KCAS

ROTOR AND ENGINE SPEED LIMITS (Steady State)

Power on:

Engine rpm	6600 to 6400
Rotor rpm	324 to 314

Power off:	
Rotor rpm transient lower limit	304 to 339
Rotor rpm	250
Power on during dives and maneuvers:	
Rotor rpm	319 to 324

TEMPERATURE AND PRESSURE LIMITS

Engine oil temperature	93°C
Transmission oil temperature	110°C
Engine oil pressure	25 to 100 psi
Transmission oil pressure	30 to 70 psi
Fuel pressure	5 to 20 psi

T53-L-13 ENGINE LIMITS

Normal rated (maximum continuous)	625°C
Military rated (30-minute limit)	645°C
Starting and acceleration (5-second limit)	675°C
Maximum for starting and acceleration	760°C
Torque pressure limit	50 psi

APPENDIX V. TEST TECHNIQUES & DATA REDUCTION PROCEDURES

INTRODUCTION

Nondimensional Method

1. Level-flight helicopter performance results may be generalized through use of nondimensional coefficients. Test results obtained at specific test conditions may be used to define accurately, performance at conditions not specifically tested. The following nondimensional coefficients were used to generalize the level-flight test results obtained during this flight test program.

$$\text{Power Coefficient} = C_P = \frac{550\ SHP}{\rho A\ (\Omega R)^3}$$

$$\text{Thrust Coefficient} = C_T = \frac{WT}{\rho A\ (\Omega R)^2}$$

$$\text{Tip-Speed Ratio} = \mu = \frac{1.689\ V_T}{\Omega R}$$

Instrumentation

2. All instrumentation was calibrated prior to being installed in the aircraft. A detailed tabulation of the instrumentation used is given in appendix III. All quantitative data obtained during this flight test program were derived from special sensitive instrumentation. Data were obtained from four sources:

a. Oscillograph.

b. Photopanel.

c. Pilot's panel (hand recorded).

d. Engineer's panel (hand recorded).

Weight and Balance

3. A high degree of control was maintained on weight and balance of the test helicopter. Variations in empty gross weight and cg

because of changes in instrumentation of helicopter components were defined by periodically weighing the helicopter. Fuel load was defined by measuring the fuel specific gravity and temperature after each fueling, and by using an external sight gage on the calibrated fuel cell to determine fuel volume. Fuel used in flight was recorded by a calibrated fuel-used system, and the results were cross-checked with the sight gage reading following each flight. Helicopter loading and cg were controlled by using ballast.

PERFORMANCE TEST

Level Flight

4. Level flight performance was defined by measuring the shaft horsepower required to maintain level flight throughout the airspeed range of the helicopter. A constant C_T was maintained by increasing altitude as fuel was consumed. A broad range of thrust coefficients was flown in the hog configuration. One speed power was flown in each of the other wing armament configurations. The results of the hog configuration level-flight tests were converted to nondimensional form and carpet-plotted as C_P versus C_T with lines of constant tip-speed ratio (μ). This carpet plot defined level flight performance for all gross weights, density altitudes and airspeeds throughout the range of C_T tested. The level-flight performance data for the other wing armament configurations were compared to the clean configuration for increase in equivalent flat plate area (f):

$$\Delta f = \frac{2\ \Delta C_P\ A\ (\Omega R)^3}{(V_T \times 1.689)^3}$$

$$\Delta f = \frac{2\ \Delta C_P\ A}{(\mu)^3}$$

5. Specific range performance was calculated from the true airspeed at a power setting and the engine fuel flow at that power setting.

$$\text{Specific Range} = \frac{\text{true airspeed}}{\text{fuel flow}} = \begin{array}{l}\text{nautical air miles}\\ \text{per pound of fuel}\end{array}$$

Autorotation

6. Autorotational descent performance data were acquired during sawtooth autorotations. Variation in rate of descent with airspeed was defined by stabilizing at a constant airspeed with a rotor speed of 324 rpm and measuring rate of descent. To determine the effect of rotor speed upon rate of descent, airspeed was stabilized and rotor speed was varied. The observed rate of descent was corrected to tapeline rate of descent by the expression:

$$R/D_{tapeline} = (dhp/dt)(T_a/T_{std})$$

Power Determination

7. The engine torquemeter is essentially a piston (restrained by oil) the pressure of which is proportional to the power output of the engine. The equation for determining the test shp as obtained from engine manufacturer test cell calibration curves is developed as outlined in paragraphs 8 through 12.

8. The horsepower transmitted by a rotating shaft may be expressed in the following manner:

$$SHP = \frac{2\pi}{12 \times 33{,}000} \times N_E \times TRQ$$

9. Calibration of the engine torquemeter system indicated that the engine shaft output torque was slightly nonlinear as a function of indicated torque pressure. This nonlinear relationship is graphically presented in figure I. This plot was used to obtain engine output torque (TRQ) by the graph with engine output torque pressure (P).

10. The rotor speed can be determined from engine output shaft speed as follows:

$$N_R = \frac{N_E}{20.383}$$

11. Substituting the last equations, a convenient equation for determining output shaft horsepower can be developed:

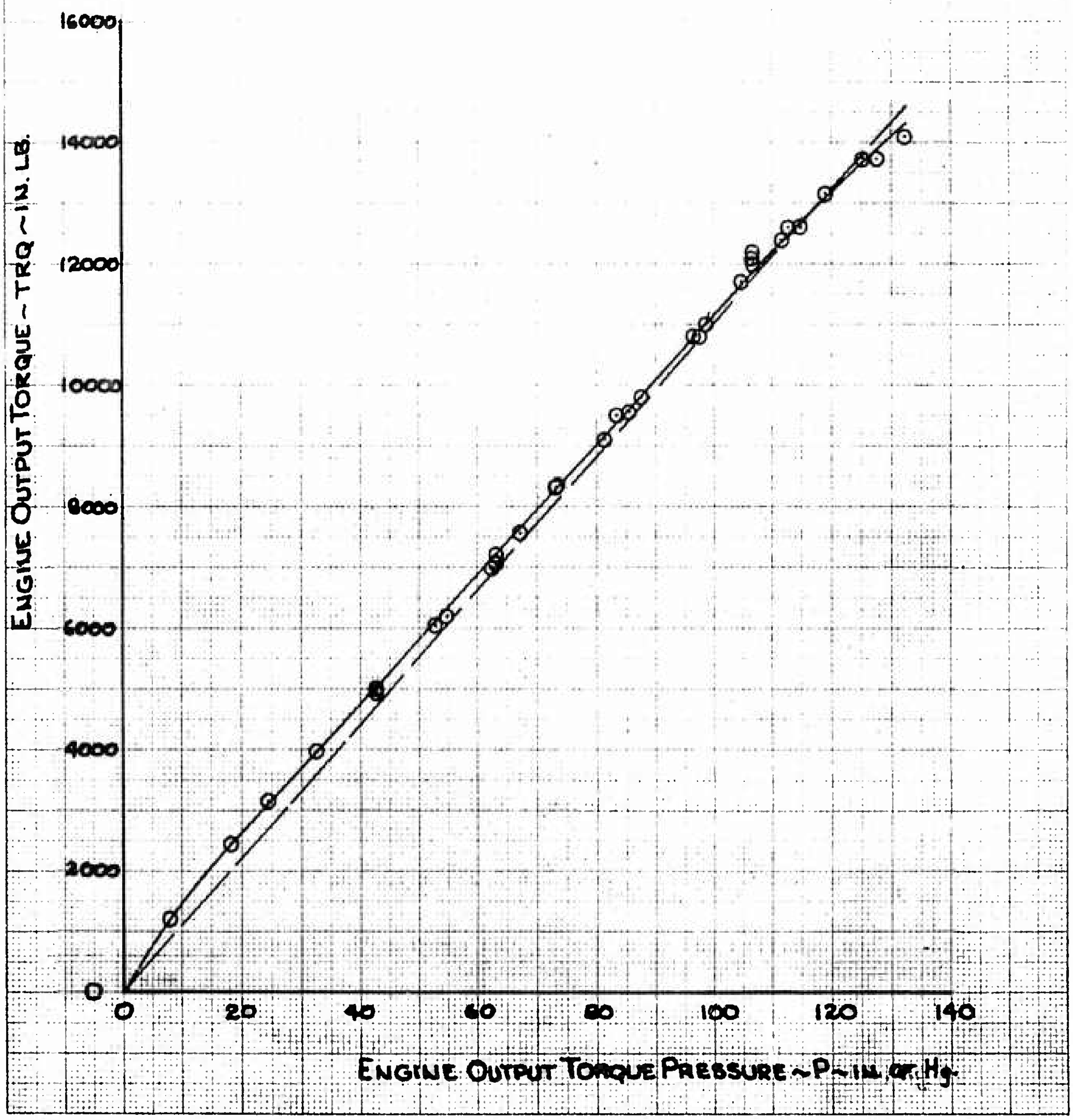
$$SHP = \frac{2\pi \times 20.383 \times TRQ \times N_R}{12 \times 33{,}000} = 3.234 \times 10^{-4} \times TRQ \times N_R$$

12. This equation was used during the program to determine the shaft horsepower for each test condition.

FIGURE NO. II
ENGINE CHARACTERISTICS
T53-L-13 S/N LE14001

NOTES: 1. POINTS OBTAINED FROM ENGINE MANUFACTURES CALIBRATION TEST CONDUCTED ON 22 AUGUST 1967

2. DASHED LINE OBTAINED FROM LYCOMING T-53-L-13 ENGINE MODEL SPECIFICATION NO. 104.33



STABILITY AND CONTROL

Directional Control Evaluation

13. Directional control tests were conducted by stabilizing the helicopter at various azimuths and airspeeds and by recording the required control positions to maintain the desired heading. A ground vehicle with a calibrated speedometer was used as an aid in stabilizing the helicopter. Ambient wind velocity and direction were measured by vane and anemometer at a location free from the effect of rotor downwash. Tests were conducted when wind velocities were less than 4 knots.

MISCELLANEOUS

Engine "Beep" Control Characteristics

14. The engine "beep" control characteristics were defined both with a loaded and unloaded main rotor system. The engine "beep" control characteristics were defined by stabilizing at a rotor speed of 324 rpm while in level flight and on the ground. The engine "beep" control was then actuated for a specified amount of time. A continuous record was made of engine and rotor speed response during the maneuver. This process was repeated until the entire speed-range authority of the "beep" control was determined.

Airspeed Calibration

15. The test airspeed indicator system (boom) was calibrated by comparing its readings to a known reference. A calibrated trailing bomb was suspended from the helicopter with a cable approximately 50 feet in length to avoid proximity effect. The aircraft was then stabilized at various airspeeds in level flight, climb and autorotation. By comparing the airspeed corrected for instrument errors of the boom system to the bomb system, the error was defined.

16. The test airspeed indicator system (boom) was calibrated at higher airspeeds, in level flight and dive using a T-28 pacer aircraft. The test and pacer aircraft were stabilized at the same airspeed, and data were recorded in each aircraft simultaneously. Since the position error of the pacer is known, the calibrated airspeed of the aircraft can be readily computed.

APPENDIX VI. SYMBOLS & ABBREVIATIONS

Listed and defined in the following table are symbols and abbreviations used in this report.

Symbols and Abbreviations	Definition	Units
A	Rotor disc area	ft^2
CG, cg	Center of gravity	---
C_P	Power coefficient	---
C_T	Thrust coefficient	---
DEG, deg	Degrees	degrees
dhp/dt	Rate of descent	ft/min
EGT	Engine exhaust gas temperature	°C
f	Equivalent flat plate area	ft^2
fig., figs.	Figure, figures	---
ft	Feet	feet
FS	Fuselage Station	inches
fwd	Forward	---
GRWT, grwt	Gross weight	pounds
H_D	Density altitude	feet
IGE	In ground effect	---
in.	Inch, inches	inches
KCAS	Knots calibrated airspeed	knots
KIAS	Knots indicated airspeed	knots
KTAS	Knots true airspeed	knots

Symbols and Abbreviations	Definition	Units
LB, lb	Weight	pounds
MAX, max	Maximum	---
MIN, min	Minimum	---
NAMPP	Nautical air miles per pound of fuel	---
N_E	Engine speed	rpm
N_R	Main rotor speed	rpm
N_1	Engine compressor speed	percent
P	Engine output torque pressure	in. of Hg
psi	Pounds per square inch	lb/in^2
R	Rotor radius	feet
R/D	Rate of descent	ft/min
ref	Reference or referred	---
RPM, rpm	Revolutions per minute	rpm
SCAS	Stability control augmentation system	---
SEC	Second	---
SHP, shp	Shaft horsepower	---
S/N	Serial number	---
STD, std	Standard	---
TEMP, temp	Temperature	°F or °C
TRQ	Engine output torque	in-lb
WT	Weight	pounds
V_{cal}	Calibrated airspeed	knots

Symbols and Abbreviations	Definition	Units
V_{cruise}	Cruise airspeed	knots
V_H	Maximum airspeed for level flight	knots
V_L	Limit airspeed	knots
V_T	True airspeed	knots
W_a	Engine air flow	lb/hour
W_{bl}	Engine bleed air flow	lb/hour
°C	Degrees centigrade	degrees
Δ	Difference	---
Ω	Main rotor angular velocity	radians/sec
μ	Tip-speed ratio	---
ρ	Air mass density	slugs/ft^3
θ	Temperature ratio	---
%	Percent	---

Listed below are the subscripts used in this report.

a	Ambient
std	Standard
t	Test

APPENDIX VII. PILOT RATING SCALE

CONTROLLABLE CAPABLE OF BEING CONTROLLED OR MANAGED IN CONTEXT OF MISSION, WITH AVAILABLE PILOT ATTENTION	ACCEPTABLE MAY HAVE DEFICIENCIES WHICH WARRANT IMPROVEMENT, BUT ADEQUATE FOR MISSION. PILOT COMPENSATION, IF REQUIRED TO ACHIEVE ACCEPTABLE PERFORMANCE, IS FEASIBLE.	SATISFACTORY MEETS ALL REQUIREMENTS AND EXPECTATIONS, GOOD ENOUGH WITHOUT IMPROVEMENT CLEARLY ADEQUATE FOR MISSION.	EXCELLENT, HIGHLY DESIRABLE	A1
			GOOD, PLEASANT, WELL BEHAVED	A2
			FAIR. SOME MILDLY UNPLEASANT CHARACTERISTICS. GOOD ENOUGH FOR MISSION WITHOUT IMPROVEMENT.	A3
		UNSATISFACTORY RELUCTANTLY ACCEPTABLE. DEFICIENCIES WHICH WARRANT IMPROVEMENT. PERFORMANCE ADEQUATE FOR MISSION WITH FEASIBLE PILOT COMPENSATION.	SOME MINOR BUT ANNOYING DEFICIENCIES. IMPROVEMENT IS REQUESTED. EFFECT ON PERFORMANCE IS EASILY COMPENSATED FOR BY PILOT.	A4
			MODERATELY OBJECTIONABLE DEFICIENCIES. IMPROVEMENT IS NEEDED. REASONABLE PERFORMANCE REQUIRES CONSIDERABLE PILOT COMPENSATION.	A5
			VERY OBJECTIONABLE DEFICIENCIES. MAJOR IMPROVEMENTS ARE NEEDED. REQUIRES BEST AVAILABLE PILOT COMPENSATION TO ACHIEVE ACCEPTABLE PERFORMANCE.	A6
	UNACCEPTABLE DEFICIENCIES WHICH REQUIRE MANDATORY IMPROVEMENT. INADEQUATE PERFORMANCE FOR MISSION EVEN WITH MAXIMUM FEASIBLE PILOT COMPENSATION.		MAJOR DEFICIENCIES WHICH REQUIRE MANDATORY IMPROVEMENT FOR ACCEPTANCE. CONTROLLABLE. PERFORMANCE INADEQUATE FOR MISSION, OR PILOT COMPENSATION REQUIRED FOR MINIMUM ACCEPTABLE PERFORMANCE IN MISSION IS TOO HIGH.	U7
			CONTROLLABLE WITH DIFFICULTY. REQUIRES SUBSTANTIAL PILOT SKILL AND ATTENTION TO RETAIN CONTROL AND CONTINUE MISSION.	U8
			MARGINALLY CONTROLLABLE IN MISSION. REQUIRES MAXIMUM AVAILABLE PILOT SKILL AND ATTENTION TO RETAIN CONTROL.	U9
UNCONTROLLABLE CONTROL WILL BE LOST DURING SOME PORTION OF MISSION.			UNCONTROLLABLE IN MISSION.	10

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13. ABSTRACT

The AH-1G helicopter Phase B, Part 6 test program was conducted at Shafter, California, and Edwards Air Force Base, California, from 12 March through 3 May 1968 by the US Army Aviation Systems Test Activity, Edwards Air Force Base, California. The program was conducted to determine level flight performance, autorotational performance, engine characteristics, armed helicopter mission capability and to evaluate the in-ground-effect (IGE) handling qualities with the canopy doors removed. The helicopter is directionally unstable when hovering IGE with either the doors on or off in winds of 9 to 13 knots for azimuth range from 160 to 260 degrees (clockwise from nose of aircraft). This instability is a major deficiency and detracts from the mission capability of the aircraft. Undue pilot attention is required to avoid overtorquing the main transmission during maneuvers requiring abrupt left-lateral cyclic inputs in forward flight. This overtorque condition will only occur below the critical altitude of the engine. Additional deficiencies and shortcomings have been published in previous reports. Sufficient performance data were not obtained to determine the guarantee compliance.

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14. KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
AH-1G helicopter Phase B, Part 6 Conducted to determine Level flight performance Autorotational performance Engine characteristics Armed helicopter Mission capability Handling qualities Canopy doors removed Directionally unstable Hovering IGE Overtorquing main transmission						

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